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ATLAS-AGENA PERFORMANCE FOR THE 1967 MARINER VENUS MISSION

Lewis Research Center
Cleveland, Ohio

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D. C. • JUNE 1969

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By Lewis Research Center Staff

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ABSTRACT

The Atlas-Agena successfully launched a Mariner unmanned deep space probe during 1967 to fly by the planet Venus. This report discusses the performance of the Atlas-Agena from lift-off through the Agena posigrade maneuver. The objective of the mission was to extend scientific knowledge about the size and mass of Venus and the density and composition of the planet's atmosphere.

CONTENTS

I. SUMMARY

This report evaluates the flight performance of the Atlas-Agena launch vehicle in support of the Mariner 67 flyby of the planet Venus. The Atlas-Agena launch accurately placed the Mariner into a Venus transfer ellipse and the Mariner passed within 3700 kilometers of the planet Venus on October 19, 1967.

The Mariner 67 space vehicle was successfully launched from the Eastern Test Range on June 14, 1967. The Atlas boosted the Agena-Mariner into the proper suborbital coast ellipse and separation of the Agena-Mariner from the Atlas was successful. The Agena first burn placed the Agena-Mariner into the desired parking orbit. After an orbital coast period, the Agena second burn placed the Agena-Mariner into the desired Venus transfer ellipse. The Mariner was separated from the Agena. The Agena then performed the required posigrade maneuver to ensure the Agena would not interfere with Mariner or impact the planet Venus.

All Atlas systems performed satisfactorily. The Mod III radio guidance system performed satisfactorily although a backup method was used by the ground station to acquire the vehicle. All Agena systems performed satisfactorily except the Agena engine experienced chamber pressure variations during second burn. These pressure variations had no effect on the attainment of the mission objective.

II. INTRODUCTION

by Roy K. Hackbarth

The purpose of the Mariner Venus 67 mission was to obtain scientific data on the physical characteristics of the planet Venus and the composition of the Venusian atmosphere. The objective of the launch vehicle was to inject the Mariner into a prescribed Venus transfer trajectory and, after spacecraft separation, to perform a posigrade maneuver to ensure the Agena would not interfere with the Mariner or impact Venus.

This was the fifth Atlas-Agena launch in support of Mariner flybys of Mars and Venus and the third launched under the direction of the Lewis Research Center.

Mariners I and II were launched toward Venus in 1962. Mariner I was unsuccessful due to a booster malfunction and Mariner II performed the first successful flyby of Venus in December 1962. Mariners III and IV were launched towards Mars in November 1964. Mariner III was unsuccessful due to failure of the nose fairing to separate and Mariner IV performed a successful flyby of Mars in July 1965. Mariner V (Mariner Venus 67) was launched toward Venus on June 14, 1967.

This report evaluates the flight performance of the Atlas-Agena launch vehicle for Mariner from lift-off through the Agena posigrade maneuver.

III. LAUNCH VEHICLE DESCRIPTION

by Roy K. Hackbarth and Eugene E. Coffey

The Atlas-Agena is a two-stage launch vehicle consisting of an Atlas first stage and an Agena second stage connected by a booster adapter. The composite vehicle (fig. III-1) including the spacecraft shroud and booster adapter is 31.7 meters (104 ft) in length. The vehicle weight at lift-off is approximately 125 600 kilograms (277 000 lb). Figure III-2 shows the Atlas-Agena launch vehicle lifting off with the Mariner Venus 67 spacecraft.

The Atlas SLV-3 (fig. III-3) is 21.34 meters (70 ft) long and is 3.05 meters (10 ft) in diameter except for the forward section of the tank which is conical and tapers to a diameter of about 2 meters (6 ft). Atlas is propelled by a standard Rocketdyne MA-5 propulsion system consisting of a booster engine having two thrust chambers with a total thrust at sea level of 1467.9×10^3 newtons (330 000 lb); a sustainer engine with a thrust at sea level of 253.55×10^3 newtons (57 000 lb); and two vernier engines, each with a thrust at sea level of 2.98×10^3 newtons (669 lb). All engines use liquid oxygen and high-grade kerosene propellants and are ignited prior to lift-off. The booster thrust chambers are gimbaled for pitch, yaw, and roll control during the booster phase of flight. This phase is completed when the vehicle acceleration equals about 6 g's. The booster engines are jettisoned about 3 seconds after booster engine shutdown. The sustainer and vernier engines continue to burn for the sustainer phase of flight. During this phase, the sustainer engine is gimbaled for pitch and yaw control, and vernier engines are gimbaled for roll control. The sustainer engine burns until the vehicle achieves the desired suborbital parameters as determined by the ground radio guidance system. After sustainer engine shutdown, the vernier engines continue to burn for a short period prior to the Atlas-Agena separation. During this phase, the vernier engines are gimbaled to provide vehicle attitude control and fine trajectory corrections. After vernier engine shutdown, the Atlas is severed from the Agena by the firing of a Mild Detonating Fuse (MDF) system located on the booster adapter. The firing of a retrorocket system, mounted on the booster adapter, then separates the Atlas booster adapter from the Agena.

The second-stage Agena and the shroud protecting the spacecraft are shown in figure III-4. The diameter of the Agena is 1.52 meters (5 ft), and the length of the Agena shroud is about 10 meters (34 ft). Agena is powered by a model 8096 Bell Aerosystems engine with a rated vacuum thrust of 71.17×10^3 newtons (16 000 lb) and has a two-burn capability. This engine uses unsymmetrical dimethylhydrazine and inhibited red fuming nitric acid as propellants. During powered flight, pitch and yaw control are provided by gimbaling the Agena engine, and roll control is provided by a cold gas (mixture of nitrogen and tetrafluoromethane) system. During periods of nonpowered flight, pitch, yaw, and roll control are provided by the cold gas system. The cold gas system and a posigrade rocket on the Agena are used to complete a posigrade maneuver after spacecraft release. A metal shroud is used to provide environmental protection for the spacecraft during ascent. This shroud is jettisoned after Atlas vernier engine shutdown just prior to Atlas-Agena separation.

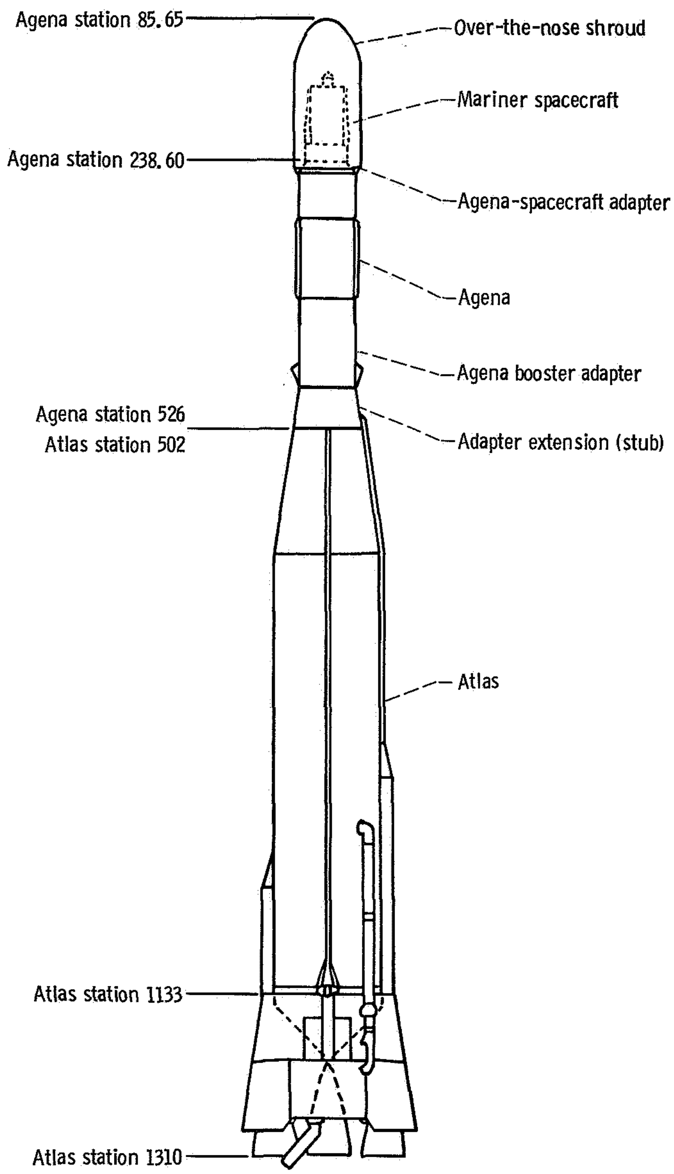


Figure III-1. - Space vehicle profile, Mariner Venus 67.

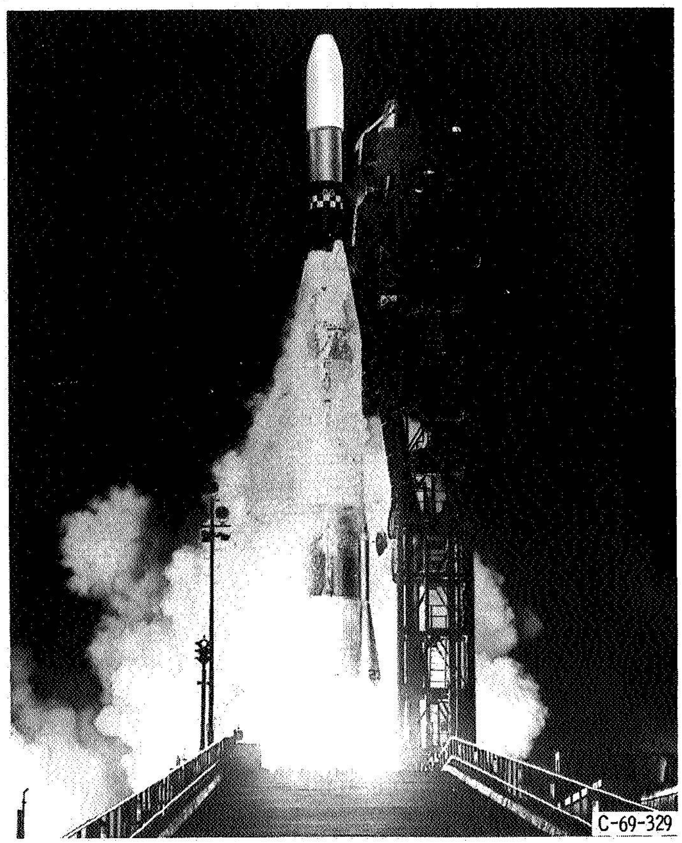


Figure III-2. - Atlas-Agena vehicle lifting off with Mariner Venus 67.

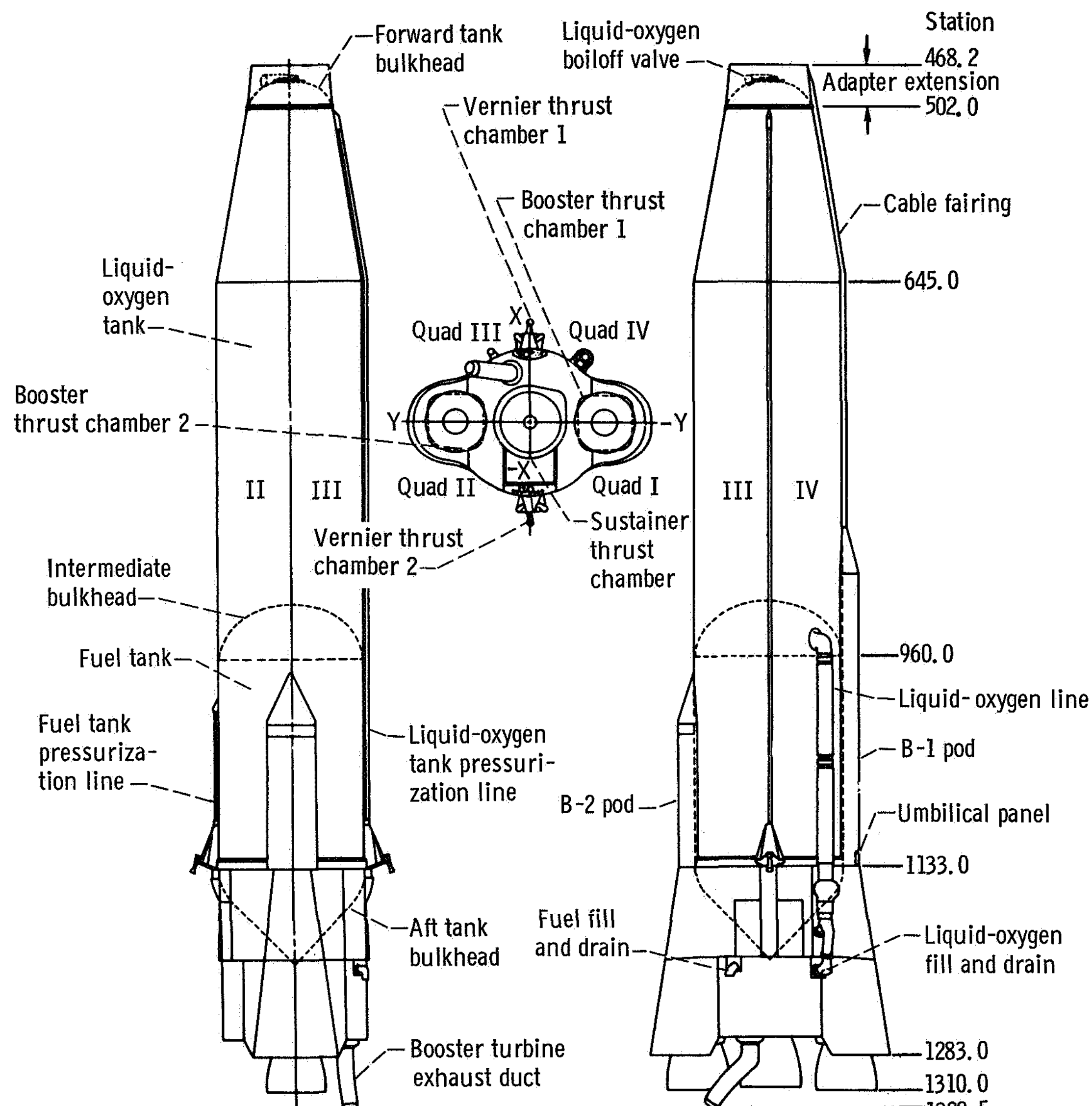


Figure III-3. - Atlas SLV-3 configuration.

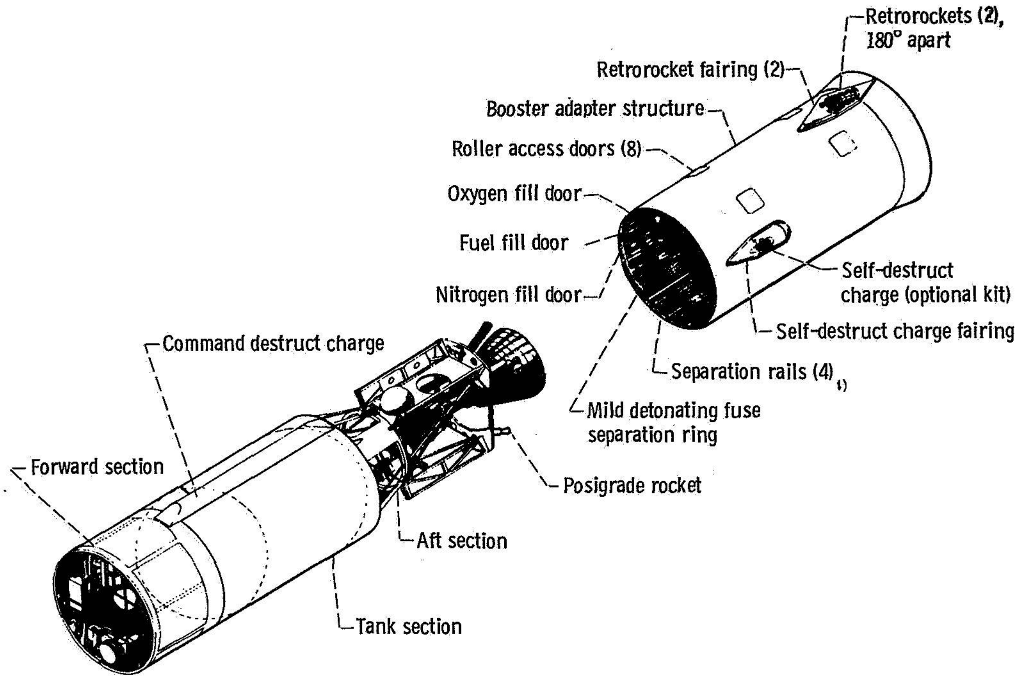


Figure III-4. - Agena configuration, Mariner Venus 67.

IV. TRAJECTORY AND PERFORMANCE

by James C. Stoll and Kenneth A. Adams

TRAJECTORY PLAN

The Atlas boosts the Agena-Mariner onto a prescribed suborbital coast ellipse, and the Agena performs two engine burns to place the Mariner onto a Venus transfer orbit. The Atlas flight consists of three powered phases: a booster engine phase, a sustainer engine phase, and a vernier engine phase. Shroud ejection, followed by Agena separation, occurs after the vernier phase is completed. Following Atlas-Agena separation, the Agena engine first burn places the Agena-Mariner onto a circular parking orbit at an altitude of 185.2 kilometers (100 n. mi.). The Agena-Mariner coasts in the parking orbit to a predetermined point where the Agena engine is reignited. The Agena engine second burn places the Agena-Mariner onto a Venus transfer orbit with the required energy to intercept the planet. The Mariner is then separated from the Agena and the Agena performs a posigrade maneuver to ensure that it will neither interfere with the spacecraft nor impact Venus.

Lunar and planetary missions require a flight azimuth which is a function of lift-off time. In practice, flight azimuths are precalculated for the midpoints of successive 15-minute intervals throughout the daily launch window. These intervals are identified as "Launch Plans." The precalculated flight azimuth for the designated launch plan is programmed in the Atlas airborne flight control programmer. The programmer commands the vehicle to roll from the pad azimuth to the precalculated flight azimuth during the 13 seconds of flight starting at lift-off plus 2 seconds (T + 2). If launch occurs at a time other than the instant corresponding to the programmed flight azimuth, a discrepancy exists between the programmed flight azimuth and the desired flight azimuth. This error requires a guidance correction during ascent. The correction, generated by the ground radio guidance computer, is accomplished by yaw steering during the sustainer phase of the Atlas flight.

TRAJECTORY RESULTS

Lift-Off Through Atlas Booster Phase

Mariner Venus 67 (MV-67) was successfully launched from Complex 12, Eastern Test Range, on June 14, 1967 at 0101:00.176 Eastern Standard Time using Launch Plan 14C. This launch plan required a programmed flight azimuth of 102.3°. The launch pad azimuth is 105.18°. A comparison of actual and expected times for the significant flight events for this mission is presented in table IV-I and in appendix A.

Winds at launch were predominately from the east from the surface to about 3048 meters (10 000 ft) altitude. Between 3048 and 6096 meters (10 000 and 20 000 ft) altitude, the winds were from the northeast and above 6096 meters (20 000 ft) from the northwest. The winds were light and had only a minor effect on the vehicle flight path. Above 6096 meters (20 000 ft) altitude, the winds were tail winds and tended to depress the trajectory slightly. A maximum wind velocity of 18.3 meters per second (60 ft/sec) from the northeast occurred at an altitude of 14 674.9 meters (48 146 ft). Wind data are shown in figure IV-1. Abrupt changes in wind velocity at altitudes between 9144 and 15 240 meters (30 000 and 50 000 ft) produced strong wind shears.

The maximum vehicle bending response was calculated to be 43.9 percent of the critical value at the Mariner-spacecraft adapter interface (Agena Station 247) and to occur at an altitude of 9404 meters (30 853 ft). The maximum booster engine gimbal angle was calculated to be 49.1 percent of the available gimbal angle in the pitch plane and to occur at an altitude of 9475.3 meters (31 087 ft). The data used for these calculations were obtained from the T - 0 (lift-off) weather balloon.

Radar tracking data show that the vehicle flight path was slightly lower than the expected trajectory during the Atlas booster phase of the flight. This deviation resulted primarily from tail winds and an approximate 0.4° excess total pitchover command by the flight control system during the booster phase. At booster engine cutoff (BECO), the vehicle position was about 1.98 kilometers (1.07 n. mi.) below and 1.34 kilometers (0.72 n. mi.) downrange of the expected position.

The trajectory deviated in the horizontal plane only slightly from the expected trajectory, resulting in a position 0.061 kilometer (0.033 n. mi.) left at BECO. This effect was due primarily to a 0.24° excess vehicle roll during the programmed roll maneuver.

During the booster phase of flight, the capability of the flight control system to accept Mod III Radio Guidance commands was enabled at T + 80 seconds; however, pitch steering commands could only be transmitted from the ground station between T + 100 and T + 110 seconds. No booster pitch steering occurred during this interval since the velocity vector angle dispersion in the pitch plane was less than the predetermined threshold for booster steering. Radio guidance yaw steering was not programmed to be used during the booster phase of flight. BECO occurred at T + 128.6 seconds by ground radio guidance command at a vehicle longitudinal acceleration of 5.9 g's. The acceleration level at BECO was 0.1 g lower than expected but within the guidance tolerance of ±0.2 g. The booster engines were jettisoned at T + 131.6 seconds.

Atlas Sustainer Phase

The trajectory remained depressed and left of the expected trajectory during the sustainer phase. Tracking data indicate that at sustainer engine cutoff (SECO) the vehicle position was about 5.79 kilometers (3.13 n. mi.) lower than, 1.01 kilometers (0.55 n. mi.) left of, and 0.73 kilometer (0.39 n. mi.) downrange of the expected position.

Sustainer steering was initiated at T + 138.4 seconds. The initial steering commands caused the vehicle to pitch up approximately 5.7° and yaw left 2.4°. These maneuvers were made to compensate for the low trajectory and to steer the vehicle to the desired flight azimuth. No corrections were made for the cross range displacement errors accumulated during the booster phase. Therefore, these errors were in evidence at SECO. SECO was commanded by ground radio guidance at T + 296.7 seconds, 0.6 second earlier than expected.

The vehicle velocity (relative to the rotating earth) at SECO was 3.05 meters per second (10 ft/sec) higher than expected. The total Atlas performance including this velocity increment and the depressed trajectory of the vehicle was consistent with the desired energy for the expected suborbital coast ellipse. The suborbital coast ellipse parameters are given in table IV-II.

Atlas Vernier Phase

Vernier engine thrust duration after SECO was approximately 21.2 seconds. During the vernier phase pitchdown and yaw, right steering commands were issued by ground radio guidance in order to place the vehicle in the proper attitude before Atlas-Agena separation. These commands displaced the vehicle 0.5° right in yaw and 3.0° down in pitch. Vernier engine cutoff (VECO) occurred by ground radio guidance command at T + 317.9 seconds, 0.5 second later than expected. Atlas insertion velocities at VECO are given in table IV-III.

Shroud separation was commanded by radio guidance at T + 320.1 seconds, 0.6 second later than expected. This delay was due to a guidance equation requirement that the transmission of this discrete must occur during the ground computer computation cycle following VECO plus 2.0 seconds.

Atlas-Agena separation occurred at T + 322.3 seconds, which was 0.8 second later than expected but consistent with the guidance equation requirement that the discrete must occur during the ground computer computation cycle following VECO plus 4.0 seconds.

Agena Engine First-Burn Phase

After Atlas-Agena separation, the Agena programmed pitchdown maneuver was initiated to place the vehicle in the proper attitude for Agena engine first burn. The start Agena primary timer (SAT) discrete had been transmitted during the Atlas phase by the ground radio guidance system at T + 308.3 seconds, 9.6 seconds later than expected. The ground guidance system had determined that the Agena would be injected onto the suborbital coast ellipse at an altitude lower than expected and, therefore, the Agena would reach first-burn altitude later than predicted. The guidance system adjusted start Agena primary timer (SAT) by 9.6 seconds so that Agena engine first burn would occur at the proper altitude.

Agena engine first ignition occurred at T + 380.4 seconds. This event and all following timer events are consistent with the 9.6 seconds adjusted SAT within the ±0.2-second timer tolerance. Thrust duration (measured from 90 percent chamber pressure to velocity meter cutoff) was 143.7 seconds, which compares favorably with the predicted burn time of 143.3 seconds. Velocity meter shutdown indicated that the proper velocity had been gained. Thrust decay added 2.82 meters per second (9.25 ft/sec) velocity compared to a predicted 3.02 meters per second (9.9 ft/sec). This predicted velocity had been compensated for by the velocity meter setting. The Agena then coasted for 794.5 seconds on the nearly circular parking orbit to the proper spatial position for the Agena engine second-burn phase. The actual parking orbit parameters are listed in table IV-IV.

Agena Engine Second-Burn Phase

The start restart timer (SRT) discrete had been generated during the Atlas phase at T + 281.7 seconds, 1.6 seconds later than predicted. The time of SRT was adjusted 1.6 seconds by the ground guidance system so that Agena engine second burn would occur at the proper spatial position.

Agena engine second ignition occurred at T + 1319.8. This event and all following timer events are consistent with the 1.6 seconds adjusted SRT within the ±0.2-second timer tolerance. The engine second-burn duration (measured from 90 percent chamber pressure to velocity meter cutoff) was 94.4 seconds, 1.2 seconds shorter than expected. During the second burn, a momentary decrease in chamber pressure occurred. The pressure returned to a value slightly greater than the initial value. The overall effect of the chamber pressure variation was to increase engine average thrust, hence, the shorter burn duration. Velocity meter cutoff indicated that the proper velocity had been gained. Thrust decay velocity was 13.32 meters per second (43.7 ft/sec) as compared to a predicted decay velocity of 15.12 meters per second (49.6 ft/sec).

Post-Second-Burn Phase

The spacecraft was separated from the Agena at T + 1576.7 seconds. The spacecraft trajectory parameters at final injection are given in table IV-V. Target point parameters for the spacecraft are given in table IV-VI.

The Agena posigrade maneuver consisting of a yaw maneuver and a posigrade rocket firing was initiated at T + 1579.7 seconds and was successfully completed at 1892.5. Integration of the resulting Agena position and velocity vectors to Venus gave the miss distances shown in table IV-VII.

The Mariner was placed in its desired orbit and performed satisfactorily.

TABLE IV-I. - SIGNIFICANT FLIGHT EVENTS, MARINER VENUS 67

Event description	Time expected, sec	Actual time, sec
Lift-off, 0101:00:176 EST	0.0	0.0
Atlas booster cutoff	128.0	128.6
Booster engine jettison	131.0	131.6
Start Agena restart timer	280.1	281.7
Sustainer engine cutoff	297.3	296.7
Start Agena primary timer	298.7	308.3
Vernier engine cutoff	317.4	317.9
Shroud separation	319.5	320.1
Atlas-Agena separation	321.5	322.3
Agena engine first ignition	370.7	380.4
Agena engine thrust at 90 percent chamber pressure	371.9	381.6
Agena engine first cutoff	515.2	525.3
Agena engine second ignition	1318.1	1319.8
Agena engine thrust at 90 percent chamber pressure	1319.3	1320.9
Agena engine second cutoff	1414.9	1415.3
Spacecraft separation	1575.1	1576.7
Start yaw maneuver	1578.1	1579.7
Stop yaw maneuver	1587.1	1588.7
Initiate posigrade rocket	1875.1	1876.8
Posigrade rocket burnout	1891.0	1892.5

TABLE IV-II. - ATLAS SUBORBITAL COAST ELLIPSE

PARAMETERS, MARINER VENUS 67

Parameter	Units	Expected	Actual
Semimajor axis	km	4489.98	4550.79
	n. mi.	2424.40	2457.23
Semiminor axis	km	3985.08	3984.86
	n. mi.	2151.77	2151.65
Radius vector magnitude at apogee	km	6558.58	6558.53
	n. mi.	3541.35	3541.32
Inertial velocity at apogee	m/sec	5724.97	5724.82
	ft/sec	18 782.7	18 782.23
Inclination	deg	29.802	29.816
Period	min	49.72	49.91

TABLE IV-III. - ATLAS INSERTION VELOCITIES AT VERNIER ENGINE CUTOFF, MARINER VENUS 67

Parameter	Units	Expected	Actual
Velocity magnitude	m/sec	5768.9	5768.6
	ft/sec	18 926.8	18 926.0
Altitude rate	m/sec	481.98	481.04
	ft/sec	1581.3	1578.2
Lateral velocity	m/sec	0.0	1.49 right
	ft/sec	0.0	4.9 right

TABLE IV-IV. - PARKING ORBIT PARAMETERS, AGENA

Parameter	Units	Actual[a]
Apogee	km	194.5
	n. mi.	105.0
Perigee	km	181.5
	n. mi.	98.0
Period	min	88.2
Inclination	deg	29.9

[a]Security classification regulations preclude listing together actual and expected parameters of Agena.

TABLE IV-V. - SPACECRAFT TRAJECTORY

PARAMETERS AT FINAL INJECTION

Parameter	Units	Actual[a]
[b]Vis viva energy, C_3	$(km/sec)^2$	8.6159
	$(n.\ mi./sec)^2$	2.512
Radius	km	6569.53
	n. mi.	3547.26
Velocity	km/sec	10.94
	ft/sec	35 892.39
Flight path angle	deg	1.97
Inclination	deg	30.31

[a]Security classification regulations preclude listing together actual and expected parameters of Agena.

[b]Defined using geocentric radius R and inertial velocity V of spacecraft as follows:

$$C_3 = V^2 - \frac{2GM_E}{R}$$

where GM_E is Earth gravitational constant.

TABLE IV-VI. - TARGET POINT PARAMETERS, SPACECRAFT

Parameter	Units	Actual
[a] $\overline{B} \cdot \overline{T}$	km n. mi.	81 579 44 049
[a] $\overline{B} \cdot \overline{R}$	km n. mi.	-65 311 -35 265
Radius of closest approach	km n. mi.	75 781 40 918
Time elapsed from spacecraft injection to point of closest approach, T_f	hr	3069.932

[a] Encounter parameters are given in $\overline{B} \cdot \overline{T}$, $\overline{B} \cdot \overline{R}$ system which is defined as follows:

$\overline{B}$ vector directed from center of planet to incoming asymptote of approach hyperbola and perpendicular to it; thus, $\overline{B}$ represents minimum approach distance of incoming asymptote

$\overline{B}$ is resolved into components $\overline{B} \cdot \overline{T}$ and $\overline{B} \cdot \overline{R}$ where

$\overline{S}$ unit vector parallel to incoming asymptote and referenced to center of planet; $\overline{B}$ and $\overline{S}$ are perpendicular

$\overline{T}$ unit vector lying in ecliptic plane and perpendicular to S

$\overline{R}$ unit vector completing right-hand orthogonal system with $\overline{S}$ and $\overline{T}$ $(R = \overline{S} \times \overline{T})$

TABLE IV-VII. - VENUS MISS DISTANCE PARAMETERS,

AGENA (ETR C-BAND DATA)

Parameter	Units	Actual
[a]$\overline{B} \cdot \overline{T}$	km n. mi.	211 100 113 985
[a]$\overline{B} \cdot \overline{R}$	km n. mi.	-151 000 -81 533
Closest approach	km n. mi.	231 100 124 784
Time elapsed from spacecraft injection to point of closest approach, T_f	hr	2970.464

[a]Encounter parameters are given in $\overline{B} \cdot \overline{T}$, $\overline{B} \cdot \overline{R}$ system which is defined as follows:

$\overline{B}$ vector directed from center of planet to incoming asymptote of approach hyperbola and perpendicular to it; thus, $\overline{B}$ represents minimum approach distance of incoming asymptote

$\overline{B}$ is resolved into components $\overline{B} \cdot \overline{T}$ and $\overline{B} \cdot \overline{R}$ where

$\overline{S}$ unit vector parallel to incoming asymptote and referenced to center of planet; $\overline{B}$ and $\overline{S}$ are perpendicular

$\overline{T}$ unit vector lying in ecliptic plane and perpendicular to S

$\overline{R}$ unit vector completing right-hand orthogonal system with $\overline{S}$ and $\overline{T}$ ($R = \overline{S} \times \overline{T}$)

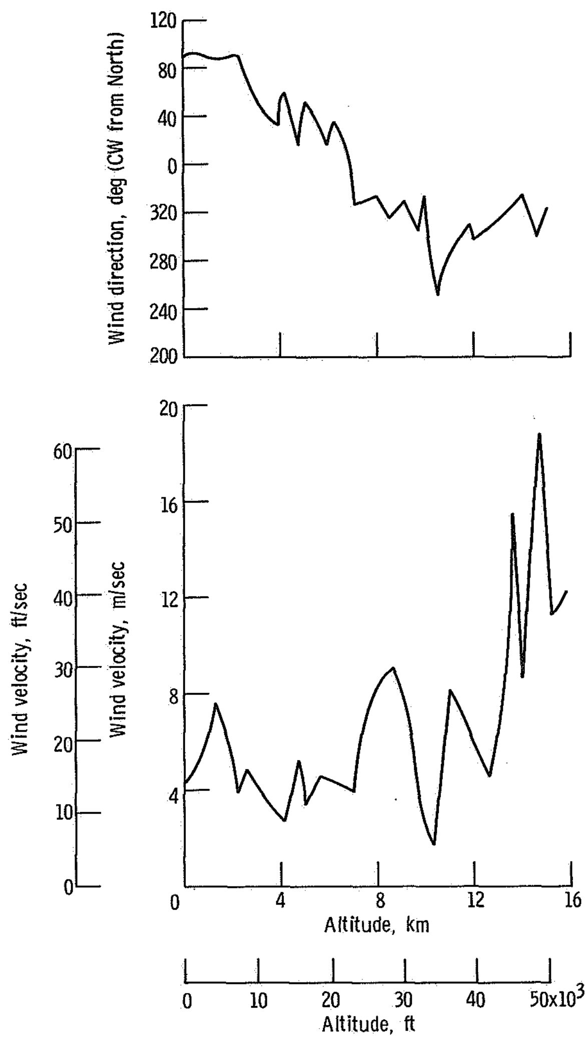


Figure IV-1. - Winds data at lift-off, Mariner Venus 67.

V. ATLAS VEHICLE SYSTEM PERFORMANCE

VEHICLE STRUCTURE SYSTEM

by Richard T. Barrett

Description

The Atlas structure system consists of two major sections: the propellant tank section and the booster engine section (fig. III-3). The propellant tank section consists of thin-walled, pressure-stabilized, stainless-steel monocoque sections of welded construction. The section is divided by a bulkhead into a fuel (kerosene) tank and an oxidizer (liquid oxygen) tank. Skin thicknesses are shown in figure V-1. The maximum allowable differential pressure between the oxidizer and fuel tanks is limited by the strength of the intermediate bulkhead. The fuel tank pressure must always be greater than the oxidizer tank pressure to prevent reversal of the intermediate bulkhead. The tank section is 3.048 meters (10 ft) in diameter and 18.5 meters (60.9 ft) in length. The forward bulkhead is ellipsoidal, and the aft bulkhead is conical. The sustainer engine is mounted on the thrust cone. Two equipment pods are attached to the sides of the tank.

The booster engine section consists of protective fairings, a thrust structure, and a booster engine with two thrust chambers. The booster engine section is attached to a thrust ring at the aft end of the fuel tank by a latch mechanism which allows the booster engine section to be jettisoned.

Performance

The vehicle structure performance was satisfactory. All measured loads were within the expected limits. The peak longitudinal load factor during Atlas flight was 5.9 g's at booster engine cutoff. The command to actuate the booster release latching mechanism was given at T + 131.6 seconds. The mechanism functioned properly, and the booster engine section jettisoned satisfactorily.

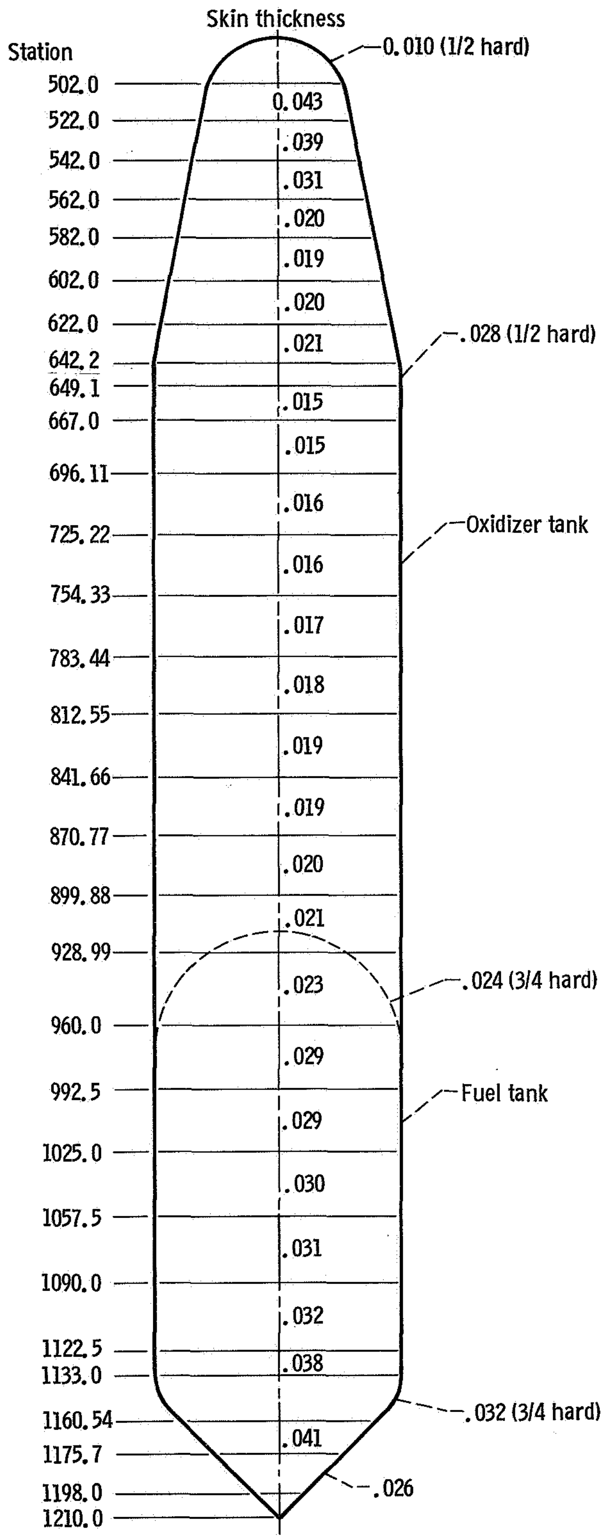


Figure V-1. - Atlas SLV-3 propellant tank section, Mariner Venus 67. (Unless noted otherwise, all material is 301 extra-full-hard stainless steel. Skin thicknesses and station numbers are in inches.)

PROPULSION SYSTEM

by Charles H. Kerrigan

Description

The Atlas engine system (fig. V-2) consists of a booster engine, a sustainer engine, two vernier engines, an engine tank system (pressurization and auxiliary propellant), and an electrical control system. The engines are of the single-burn type. During engine start, electrically fired pyrotechnic igniters are used to ignite the gas generator propellants for driving the turbopumps; and hypergolic igniters are used to ignite the propellants in the thrust chambers of the booster, sustainer, and vernier engines. The propellants are liquid oxygen and RP-1 (kerosene).

The booster engine, rated at 1468×10^3 newtons (330×10^3 lb) thrust at sea level, is made up of two gimbaled thrust chambers, propellant valves, two oxidizer and two fuel turbopumps driven by one gas generator, a lubricating oil system, and a heat exchanger. The sustainer engine, rated 253.5×10^3 newtons (57×10^3 lb) thrust at sea level, consists of a thrust chamber, propellant valves, one oxidizer and one fuel turbopump driven by a gas generator, and a lubricating oil system. The entire sustainer engine system gimbals. Each vernier engine is rated at 2.98×10^3 newtons (669 lb) thrust at sea level when supplied with propellants from the sustainer turbopumps during sustainer engine operation. In the vernier phase of flight, each vernier engine is rated at 2.34×10^3 newtons (525 lb) thrust at sea level. For this phase, the vernier engines are supplied with propellants from the engine tank system because the sustainer turbopumps do not operate after sustainer engine cutoff.

The engine tank system is composed of two small propellant tanks (each approx. 51 cm (20 in.) in diam) and a pressurization system. This system supplies propellants for starting the engines and also for vernier engine operation after sustainer engine cutoff.

Performance

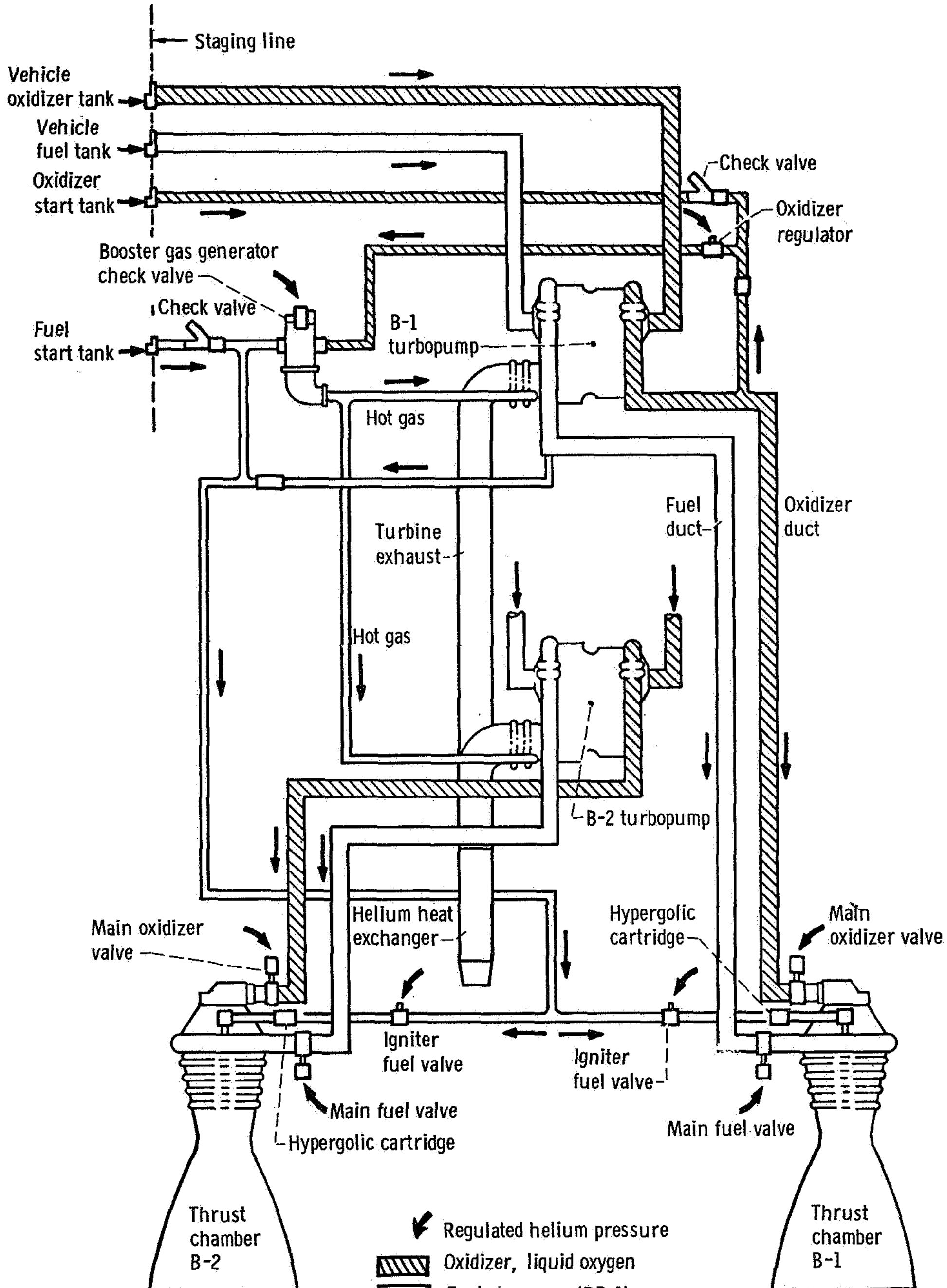
The performance of the Atlas propulsion system for the Mariner Venus 67 mission was satisfactory. During the engine start phase, valve opening times and starting sequence events were within tolerances. The flight performance of the engines was evaluated by comparing measured engine parameters with the expected values. These are tabulated in table V-I. All engine cutoff signals were issued by guidance system commands and were properly executed. Transients at engine shutdown were normal.

TABLE V-I. - ATLAS PROPULSION SYSTEM PERFORMANCE, MARINER VENUS 67

Performance parameters	Units	Expected operating range	Flight values at -			
			T + 10 sec	Booster engine cutoff	Sustainer engine cutoff	Vernier engine cutoff
Booster engine:						
Number 1 thrust chamber	N/cm^2	386 to 410	392	392	(a)	(a)
pressure	psi	560 to 595	568	568		
Number 2 thrust chamber	N/cm^2	386 to 410	398	398		
pressure	psi	560 to 595	576	576		
Gas generator chamber pressure	N/cm^2	351 to 382	358	358		
	psi	510 to 555	519	519		
Number 1 turbopump speed	rpm	6225 to 6405	6316	6286		
Number 2 turbopump speed	rpm	6165 to 6345	6202	6187		
Sustainer:						
Engine thrust chamber pressure	N/cm^2	469 to 493	483	483	483	
	psi	680 to 715	700	700	700	
Engine gas generator discharge	N/cm^2	427 to 469	446	441	441	
pressure	psi	620 to 680	648	640	640	
Engine turbopump speed	rpm	10 025 to 10 445	10 422	10 292	10 425	
Vernier:						
Engine number 1 thrust chamber	N/cm^2	172 to 183	174	173	177	
pressure when pump supplied	psi	250 to 265	252	250	256	
Engine number 1 thrust chamber	N/cm^2	145 to 155	(a)	(a)	(a)	152
pressure when tank supplied	psi	210 to 225	(a)	(a)	(a)	220
Engine number 2 thrust chamber	N/cm^2	172 to 183	177	173	177	(a)
pressure when pump supplied	psi	250 to 265	256	250	256	(a)
Engine number 2 thrust chamber	N/cm^2	145 to 155	(a)	(a)	(a)	152
pressure when tank supplied	psi	210 to 225	(a)	(a)	(a)	220

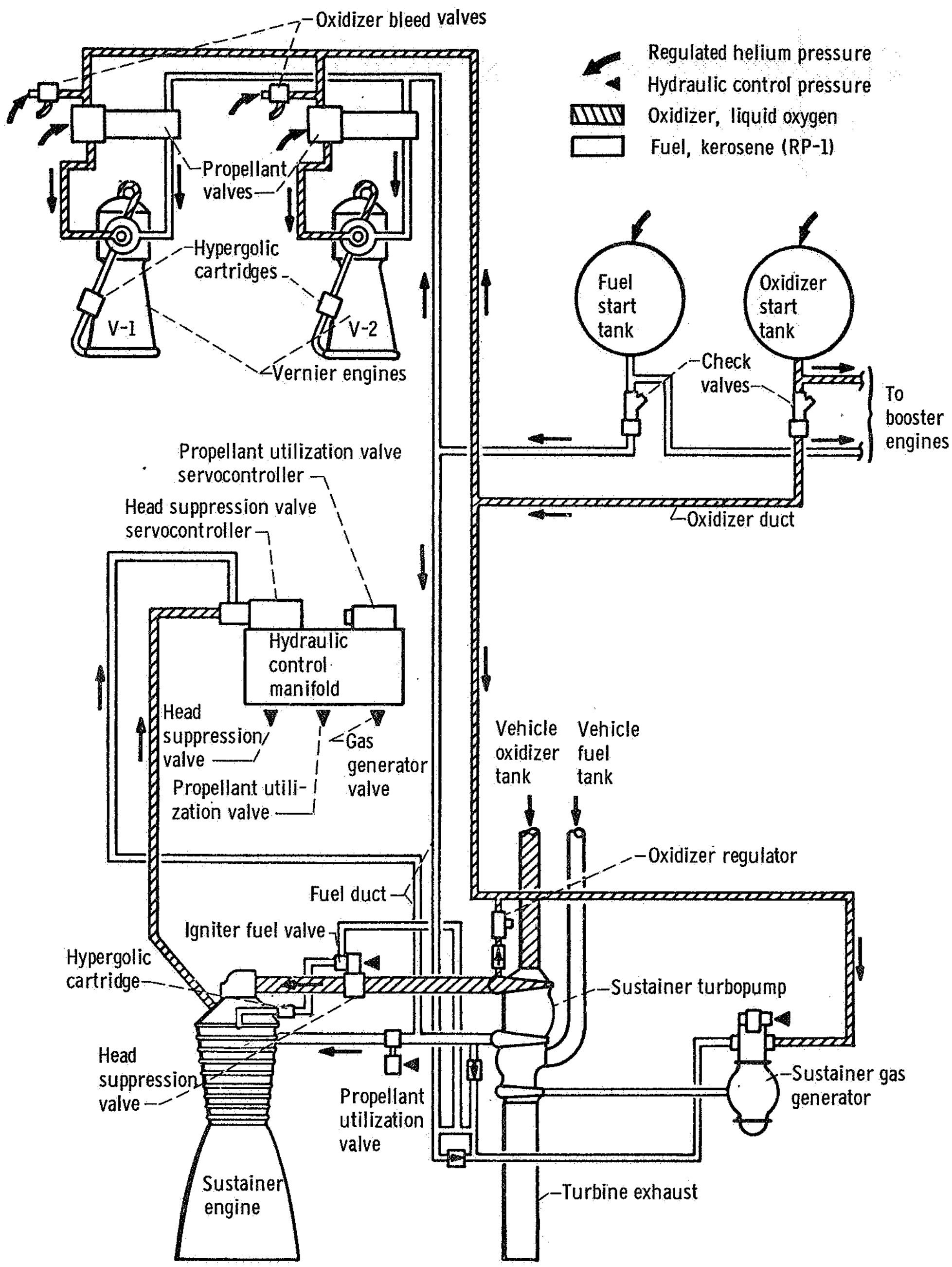
[a]Not applicable.

Performance parameter	Operating time, sec	
	Expected	Actual
Duration of booster engine burn	128.4	128.6
Duration of sustainer engine burn	295.5	296.7
Duration of vernier engine burn	315.4	317.9



(a) Booster.

Figure V-2. - Atlas propulsion system, Mariner Venus 67.



(b) Sustainer and vernier.

Figure V-2. - Concluded.

PROPELLANT UTILIZATION SYSTEM

by Clifford H. Arth

Description

The Atlas propellant utilization system (fig. V-3) is designed to cause near simultaneous depletion of both propellants. This is a digital type system which adjusts the operating mixture ratio of the sustainer engine by sampling the propellant volume ratio at six discrete points during flight. Six fuel- and six oxidizer-level sensors are positioned in the propellant tanks so that both sensors will uncover simultaneously if the propellants are being consumed at the proper ratio. If the propellant usage ratio is incorrect, one sensor of a pair will uncover before the other sensor. The time difference in the uncovering of the sensors comprising a pair is directly proportional to the propellant usage ratio error. If this time difference is greater than the limit error times for each sensor pair, the propellant utilization valve will be commanded to the full open or closed position, depending on which sensor uncovers first. If the actual error time is less than the limit error time, the valve will be commanded to something less than full open or closed position. This adjustment will theoretically result in a zero error time when the liquid level reaches the next sensor pair.

This difference in uncovering time for each sensor pair is measured and an error signal is transmitted to a hydraulic control unit. This hydraulic control unit directly controls the position of the propellant utilization (fuel) valve and indirectly controls the position of the liquid oxygen valve. When an error signal is sent to the propellant utilization valve for an increase in fuel flow, the fuel pump discharge pressure will decrease as the valve moves open. The liquid oxygen head suppression servocontrol senses this decreasing pressure and causes the liquid oxygen head suppression valve to move to restrict the flow of the liquid oxygen to the thrust chamber, thus decreasing the liquid oxygen injection pressure by approximately the same amount as the decrease in RP-1 (fuel) pump discharge pressure. The combined performance of the liquid oxygen head suppression valve and the propellant utilization system results in a near-constant total flow weight of propellants to the sustainer engine.

Performance

The burnable propellant residuals in the propellant tanks at sustainer engine cutoff were 260 kilograms (574 lb) of fuel and 207 kilograms (457 lb) of liquid oxygen. The residuals would have allowed the sustainer engine to burn an additional 2.3 seconds.

However, the proper velocity had been attained, and the guidance system shut down the engine. If the flight would have continued to theoretical liquid oxygen depletion, the total fuel remaining would have been 172 kilograms (380 lb).

Fuel and oxidizer pressure sensing ports provide the final propellant level data. The differential pressure between the ullage and the port pressure is measured. When the pressure differential indicates zero, the propellant level is below the port. As the sense port uncovers, a time interval is calculated from the instant of port uncovering to sustainer engine cutoff. This time interval combined with flow rate information is used in determining the propellant residuals. The fuel sensor port uncovering time was 2.1 seconds, and the liquid oxygen sensor port uncovering time was 6.0 seconds.

Table V-II presents the error between the fuel and liquid oxygen sensor uncovering times for all six sensor pairs. The data show that the error times are all within the limit times; thus, at no time during the flight was full correction capability necessary.

TABLE V-II. - LEVEL SENSOR ERROR TIMES,

MARINER VENUS 67

Sensor	Limit error time, sec	Actual error time, sec	First sensor uncovered	First sensor uncovering time, sec
1	1.018	1.0	Liquid oxygen	T + 7.2
2	.968	.9	Liquid oxygen	T + 46.2
3	.78	.3	Liquid oxygen	T + 83.8
4	1.89	.05	Liquid oxygen	T + 114
5	8.4	.5	Liquid oxygen	T + 192.7
6	4.2	1.0	RP-1 (kerosene)	T + 247.7

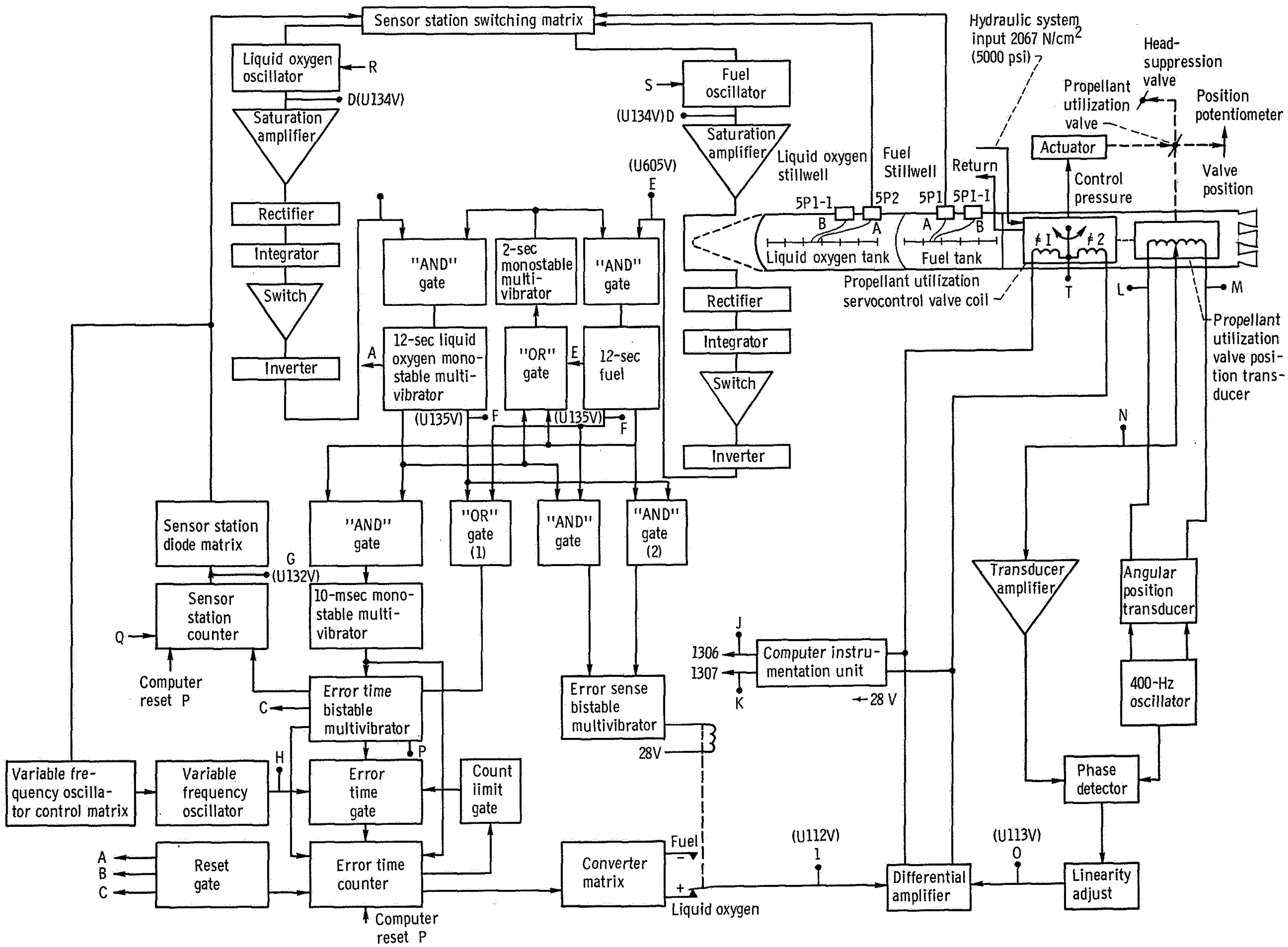


Figure V-3. - Atlas propellant utilization system, Mariner Venus 67.

HYDRAULIC SYSTEM

by Eugene J. Cieslewicz

Description

Two hydraulic systems, shown in figure V-4, are used to supply fluid power for operation of the sustainer control valves and for thrust vector control of all Atlas engines. One system is used for the booster engine and the other for the sustainer and vernier engines.

The booster hydraulic system provides power solely for gimbaling the two thrust chambers of the booster engine system. System pressure is supplied by a single, pressure-compensated, variable-displacement pump driven by the engine turbopump accessory drive. Other components of the system include four servocylinders, a high-pressure relief valve, an accumulator, and a reservoir. Engine gimbaling in response to flight control commands is accomplished by the servocylinders, which provide separate pitch, yaw, and roll control during the booster phase of flight. The maximum booster engine gimbal angle capability is $\pm 5^{\circ}$ in both the pitch and yaw planes.

The sustainer hydraulic system is similar to that of the booster. It provides hydraulic power for gimbaling the sustainer engine, for sustainer engine control valves and for gimbaling of the two vernier engines. The sustainer engine is held in the centered position until booster engine cutoff. Any disturbances created by the engine differential cutoff impulses are damped by gimbaling the sustainer and vernier engines. The sustainer engine is again centered during booster engine section jettison. Vehicle roll control is maintained throughout the sustainer phase by differential gimbaling of the vernier engines. During vernier solo operation, after sustainer engine cutoff, the vernier engines gimbal actuators are provided with hydraulic pressure from two pressurized accumulators. Actuator limit travel of the vernier engines is $\pm 70^{\circ}$, and the sustainer engine is $\pm 3^{\circ}$.

Performance

Hydraulic system pressure data for both the booster and sustainer-vernier circuits are shown in table V-III. Transfer of fluid power from ground to airborne systems before lift-off was normal. Starting transients produced a normal overshoot of about 10 percent in the hydraulic pump discharge pressures. Pressures, except for the expected transients at lift-off, booster engine cutoff, and sustainer engine cutoff, were stable throughout the Atlas flight phase. Gimbaling of the engines was well within the gimbal capabilities and in accordance with the flight control and guidance requirements.

TABLE V-III. - ATLAS HYDRAULIC SYSTEM PERFORMANCE DATA,

MARINER VENUS 67

Performance parameter	Units	Flight value at -			
		Lift-off	Booster engine cutoff	Sustainer engine cutoff	Vernier engine cutoff
Booster pump discharge pressure, absolute	N/cm^2	2148	2148	(a)	(a)
	psi	3115	3115		
Booster accumulator pressure, absolute	N/cm^2	2172	2172	(a)	(a)
	psi	3150	3150		
Sustainer pump discharge pressure, absolute	N/cm^2	2124	2124	2124	(b)
	psi	3080	3080	3080	
Sustainer-vernier pressure, absolute	N/cm^2	2082	2082	2075	772
	psi	3020	3020	3010	1120

[a]Not applicable after booster engine cutoff.

[b]Not applicable after sustainer engine cutoff.

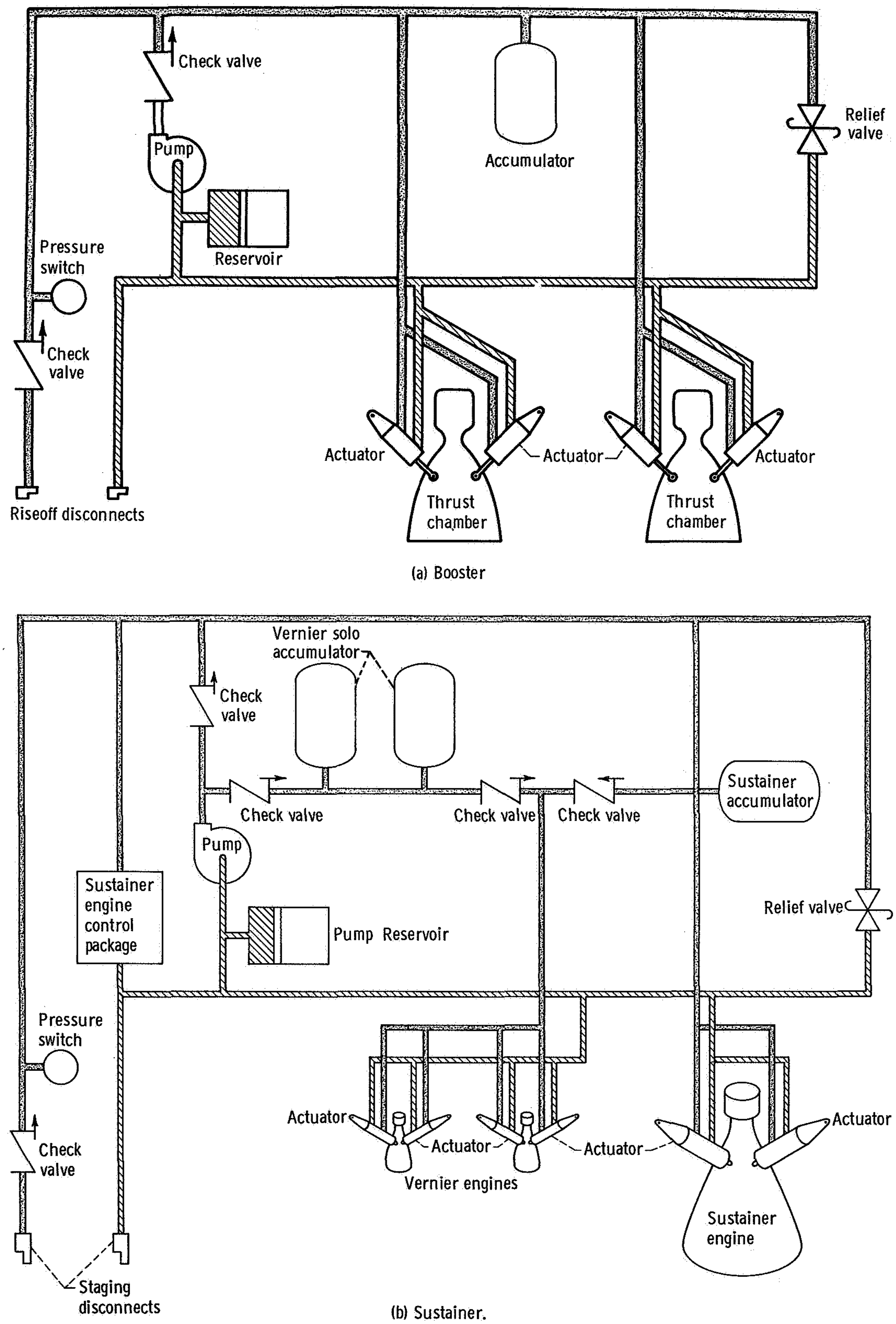


Figure V-4. - Atlas hydraulic system, Mariner Venus 67.

PNEUMATIC SYSTEM

by Eugene J. Fourney

Description

The Atlas pneumatic system supplies helium gas for tank pressurization and for various vehicle control functions. The system consists of three independent subsystems: propellant tank pressurization, engine control, and booster section jettison. The system schematic is shown in figure V-5.

The propellant tank pressurization subsystem is used to maintain propellant tank pressures at required levels to support the pressure stabilized tank structure, and to satisfy the inlet pressure requirements of the engine turbopumps. In addition, this subsystem supplies helium to pressurize the hydraulic reservoirs and turbopump lubricant storage tanks. The subsystem consists of six shrouded helium storage bottles, a heat exchanger, and fuel and oxidizer tank pressure regulators and relief valves. The six shrouded helium storage bottles with a total capacity of 724 144 cubic centimeters (44 190 cu in.) are mounted in the jettisonable booster engine section. The bottle shrouds are filled with liquid nitrogen during prelaunch operations to chill the helium in order to provide a maximum storage capacity at an absolute pressure of 2068 newtons per square centimeter (3000 psi). The liquid nitrogen drains from the shrouds at lift-off. During flight, the helium passes through a heat exchanger located in the booster engine turbine exhaust duct and is heated before being supplied to the tank pressure regulators. Control of propellant tank pressurization subsystem is switched from the ground to the airborne regulators at about T - 60 seconds. The airborne regulators are set to control fuel tank gage pressure between 44.13 and 46.2 newtons per square centimeter (64 and 67 psi) and the oxidizer tank pressure between 19.65 and 21.37 newtons per square centimeter (28.5 and 31.0 psi). Pneumatic regulation of tank pressure is terminated at booster staging. Thereafter, the fuel tank pressure decays slowly. The oxidizer tank pressure is augmented by liquid oxygen boiloff; therefore, the pressure decay in this tank is much smaller.

The engine controls subsystem supplies helium pressure for actuation of engine control valves, for pressurization of the engine start tanks, for purging booster engine turbopump seals, and for the reference pressure to the regulators which control oxidizer flow to the gas generator. Control pressure in the system is maintained through Atlas-Centaur separation. These pneumatic requirements are provided from a 76 000-cubic-centimeter (4650-cu-in.) supply bottle.

The booster engine jettison subsystem supplies pressure to release of the pneumatic staging latches for jettison of the booster engine section. A command from the Atlas flight control system opens two explosively actuated valves to supply helium pressure to the 10 piston-operated staging latches. Helium for the system is supplied by a 14 257-cubic-centimeter (870-cu-in.) bottle charged to a gage pressure of 2068 newtons per square centimeter (3000 psi).

Performance

The pneumatic system performance was satisfactory. Propellant tank pressures were satisfactory and all control functions were performed properly. Pneumatic system data are presented in table V-IV. Liquid oxygen tank ullage pressure oscillations were within the range experienced on previous flights. Prior to lift-off, oscillation frequencies of 3.25 hertz were measured. The oscillation amplitudes (differential pressure across the bulkhead) varied with a maximum peak-to-peak amplitude of 2.01 newtons per square centimeter (3.0 psi). After lift-off, these oscillations increased in frequency to 5.25 hertz and increased in amplitude slightly. These oscillations damped out within approximately the same time span as had been experienced on other similar flights. These oscillations result from the configuration of the regulator and are considered normal.

TABLE V-IV. - PNEUMATIC SYSTEM PERFORMANCE, MARINER VENUS 67

Performance parameter	Measurement number	Units	Design range	Flight values at -					Remarks
				T - 10 seconds	T - 0 seconds	Booster engine cutoff	Sustainer engine cutoff	Vernier engine cutoff	
Oxidizer tank ullage pressure, gage	AF1P	N/cm^2	19.65 to 21.37	21.83	20.1	20.48	[a]19.44	[a]19.44	---------------
		psi	28.5 to 31.0	31.8	29.3	29.7	[a]28.2	[a]28.2	---------------
Fuel tank ullage pressure, gage	AF3P	N/cm^2	44.13 to 46.20	45.71	45.02	45.3	[a]33.58	[a]33.58	---------------
		psi	64.0 to 67.0	66.3	65.3	65.7	[a]48.7	[a]48.7	---------------
Intermediate bulkhead differential pressure	AF116P	N/cm^2	0.35 (min.)	11.72	10.27	11.97	13.79	13.79	Lowest value was 5.76 N/cm^2 (8.5 psi) at T + 1 second.
		psi	0.5 (min.)	17.0	14.9	17.5	20.0	20.0	
Sustainer controls bottle pressure, absolute	AF241P	N/cm^2	2344 (max.)	2172	2072	1737	1689	1062	---------------
		psi	3400 (max.)	3150	3005	2520	2450	1540	---------------
Booster helium bottle pressure, absolute	AF246P	N/cm^2	2344 (max.)	2124	2058	434	(a)	(a)	---------------
		psi	3400 (max.)	3080	2987	630	(a)	(a)	---------------
Booster helium bottle temperature	AF247T	K	-----------	77.5	79.4	43.15	(a)	(a)	---------------
		^{o}F	-----------	-320	-317	-382	(a)	(a)	---------------

[a]Helium supply bottles are jettisoned with booster engines at booster engine cutoff +3 seconds; therefore, regulator design range is no longer a criterion.

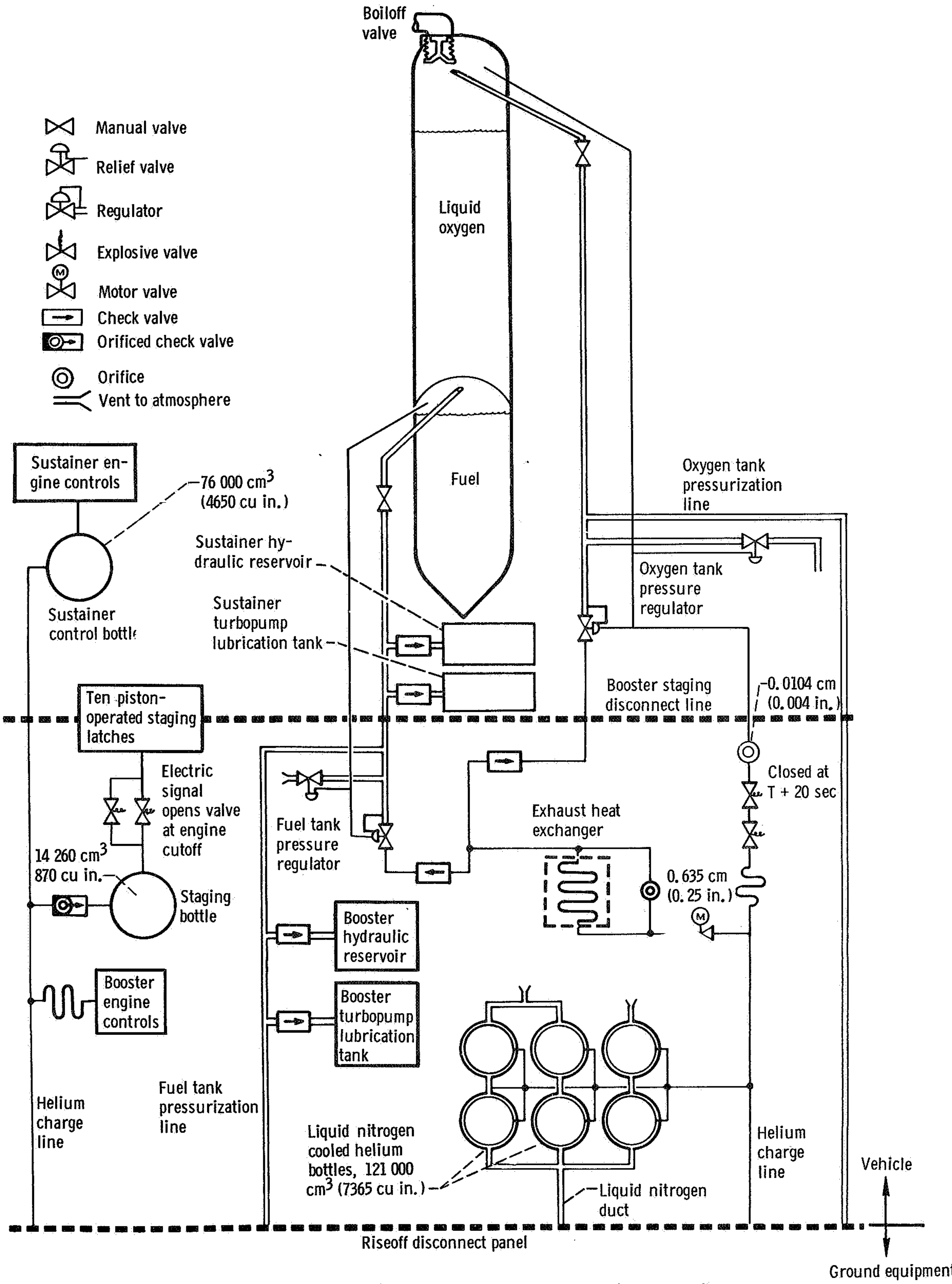


Figure V-5. - Atlas vehicle pneumatic system, Mariner Venus 67.

GUIDANCE AND FLIGHT CONTROL SYSTEM

by Dean W. Bitler and James L. Swavely

Description

The Atlas flight path is controlled by two interrelated systems: the flight control system and the Mod III Radio Guidance System. The flight control system directs the vehicle in a programmed open-loop mode from lift-off through vernier engine cutoff. During the period between T + 100 and T + 110 seconds the Mod III Radio Guidance System may generate and transmit pitch steering signals to the vehicle. During the sustainer and vernier solo phases of flight, the Mod III Radio Guidance System may generate and transmit pitch and yaw steering signals to the vehicle. The transmitted steering signals are received in the airborne guidance system and routed to the flight control system to provide corrections for vehicle deviations from the programmed trajectory.

The Mod III Radio Guidance System is the primary source for initiation of discrete commands for booster engine cutoff (BECO), sustainer engine cutoff (SECO), vernier engine cutoff (VECO), shroud separation, start Agena timer (SAT), start Agena restart timer (SRT), and Atlas-Agena separation.

The Atlas flight control system (fig. V-6) consists of the four major components described below:

(1) The displacement gyro canister contains three single-degree-of-freedom, floated, rate integrating (displacement) gyros; one single-degree-of-freedom, floated, rate gyro; and associated electronic circuitry for gain selection and signal amplification. The displacement gyros are mounted in an orthogonal triad configuration alining the input axis of each gyro to its respective vehicle axis of pitch, yaw, or roll. Each displacement gyro provides an electrical output signal proportional to the difference in angular position of the measured axis from the gyro reference axis. The input axis of the rate gyro is alined with the vehicle roll axis. The rate gyro provides an electrical output signal proportional to the angular rate of rotation of the vehicle about the gyro input (reference) axis.

(2) The rate gyro canister contains two single-degree-of-freedom, floated, rate gyros and associated electronic circuitry. The input axes of these rate gyros are alined to their respective vehicle axes of pitch and yaw. Each rate gyro provides an electrical output signal proportional to the angular rate of rotation of the vehicle about the gyro input (reference) axis.

(3) The servoamplifier canister contains electronic circuitry to amplify, filter, integrate, and algebraically sum gyro output and engine position feedback signals. The electrical outputs of this unit direct the hydraulic actuators which, in turn, gimbal the engines to provide thrust vector control.

(4) The programmer canister contains an electronic timer; arm/safe switch; high, low, and medium power electronic switches; the fixed pitch program; and circuitry to set the roll program prior to launch. The programmer issues discrete commands to other subsystems.

The Mod III Radio Guidance System includes the Atlas airborne pulse beacon, rate beacon, and decoder; and a ground station comprised of a monopulse X-band (radar) track subsystem, a continuous wave L-band rate subsystem, and a digital guidance computer subsystem. The major functions (fig. V-7) are described in the following paragraphs.

The ground track subsystem measures range, azimuth, and elevation and transmits a composite message-train containing an address code and the coded steering or discrete commands. When the address code of the received signal is correct, the airborne pulse beacon transmits a return pulse to the ground station; and the airborne decoder translates the message, and then issues steering signals or discrete commands to the flight control system. The resulting steering outputs from the decoder are 400-hertz square waves of variable phase and amplitude which are transmitted to the autopilot to torque the appropriate gyro. The gyro torque rate is proportional to the decoder output. The maximum gyro torque rate is 2 degrees per second for 100 percent steering commands.

The ground rate subsystem transmits two continuous wave signals of different frequencies from a single ground antenna. The vehicle-borne rate beacon is interrogated by the signals from the ground subsystem. The rate beacon transmits a continuous wave signal at a frequency equal to the arithmetic average of the frequencies of the received signals. This signal is received by the central rate station and two outlying rate stations. The two-way Doppler shifts and phase relations of the signals as received at these ground stations are used to determine the vehicle range, azimuth, and elevation rates.

The position and rate information from the ground track and ground rate subsystems is sent to the ground computer. The ground computer solves the guidance equations every 1/2 second using position and rate information. The ground computer then generates steering and discrete commands which are transmitted from the computer to the ground track subsystem and then to the vehicle.

The ground track subsystem conical scan antenna acquires the vehicle during an early portion of the flight. Once the vehicle is acquired by the conical scan antenna, tracking is automatically switched to the main track antenna which is on the same mount as the conical scan antenna. The ground rate subsystem antennas are electronically slaved to the main track antenna.

The primary method used to acquire the vehicle is known as cube acquisition. In the cube acquisition method, the conical scan antenna is pointed to one of seven predetermined cubes along the programmed trajectory. If the vehicle is not acquired in the first cube selected, the conical scan antenna may be automatically switched to subsequent predetermined cubes. Alternate methods of acquisition are optical tracking or slaving to real-time vehicle coordinates supplied by the Eastern Test Range.

Performance

The performance of the Atlas flight control was satisfactory. Lift-off transients in pitch, yaw, and roll were within acceptable limits. The maximum vehicle displacement angles during the lift-off transients were 0.9° clockwise in roll, 0.11° up in pitch, and 0.28° right in yaw. The roll program was initiated at T + 2 seconds and ended at T + 15 seconds as planned. The programmed vehicle roll required to achieve the desired launch azimuth was 2.88°. The actual vehicle roll from roll rate gyro data was 3.12°, which was within acceptable vehicle roll requirements.

The pitch program was initiated at T + 15 seconds as planned. The programmed times, actual times, and pitch rates for each step of the pitch program are listed in table V-V. The actual pitch maneuver dispersions were within acceptable limits.

Maximum dynamic pressure occurred at T + 72.8 seconds after lift-off. Dynamic disturbances during the period of maximum dynamic pressure were small and resulted in the booster engines gimbaling a maximum of 1.55° in pitch and 0.48° in yaw. This represented 31.0 percent of the engine capability in pitch and 9.6 percent in yaw. The gimbal angles were within the maximum gimbal angle predictions based on atmospheric data (wind soundings) taken at T - 0 (see IV. TRAJECTORY AND PERFORMANCE).

Vehicle displacements resulting from booster engine shutdown were 0.19° down in pitch and 0.51° left in yaw, and were within acceptable limits. The booster engine jettison sequence was normal and the resulting small yaw transient was quickly damped. The pitch transient resulted in an oscillation of 0.166 hertz with a peak-to-peak amplitude of 1.8 degrees per second This oscillation was reduced to within ±10 percent by T + 164.0 seconds. Sustainer steering was initiated at T + 138.4 seconds and was normal for this period of flight. The vernier solo phase of flight was normal, and steering commands during this period were within acceptable limits.

After completion of the vernier solo phase of flight, the command to separate Agena from the Atlas was initiated by radio guidance. At this time, the vehicle was stable in attitude and separation was successfully completed.

Postflight evaluation of the Mod III ground station data indicated that the equipment performed satisfactorily except for the problem with cube acquisition by the ground radar. Acquisition in the first cube at approximately T + 60 seconds was not accomplished because the preflight Cube 1 elevation setting for the ground station track antenna was in error.

At T + 60 seconds, when acquisition was expected to occur, the received signal at the ground station was abnormally low. When it became evident to the track console operator that acquisition in the first cube would not occur, he switched to the backup optical track method for acquisition. With the aid of the optical tracker, the track subsystem conical scan antenna acquired the vehicle at T + 89.7 seconds. The automatic switch to monopulse tracking with the main antenna occurred at T + 95.6 seconds, and good track data were presented to the computer by T + 98.9 seconds. The late acquisition of the vehicle by Mod III ground guidance had no detrimental effect on the mission.

A postflight investigation of the acquisition problem revealed that the manual constant setting on the guidance track console for Cube 1 elevation was set in error, which caused the track antenna to point to an elevation angle of 88.7^{o} instead of the required 48.8^{o} for Cube 1. The 88.7^{o} setting was in accordance with the data sheet supplied to the console operator; however, a tabulation error had been made during generation of the data sheet for the Mariner Venus 67 mission.

Track lock was continuous from acquisition in the monopulse mode until T + 410.2 seconds, 88.3 seconds after Atlas-Agena separation. Track lock was then intermittent until final loss of lock occurred at T + 423.6 seconds when the Atlas was at an elevation angle of 1.9^{o} above the horizon. The signal received by the track subsystem from T + 98.9 seconds until final loss of lock was within 3 decibels of the theoretically expected level.

Lock at all rate antennas was accomplished by T + 68.8 seconds, with good rate data presented to the computer by T + 70.2 seconds. After a period of 6.0 seconds of good rate lock, the signal became intermittent due to the acquisition problem discussed previously. The signal level received by the central rate ground station during this 6.0-second period averaged -104 dBm (decibel referenced to 1 MW), compared to the expected -60 to -70 dBm. Lock at all rate antennas was again accomplished at T + 89.2 seconds, and good rate data were presented to the computer by T + 90.7 seconds. Rate lock was continuous thereafter until T + 403.5 seconds. An intermittent period of lock then occurred followed by final loss of lock at T + 413.5 seconds.

The computer subsystem performance was satisfactory throughout the countdown and vehicle flight. Following the flight, the guidance program was successfully verified before removal of the program from the computer. A simulated rerun of the flight indicated that no transient errors occurred during the flight.

The pulse beacon automatic gain control (AGC) monitor indicated a received signal strength of approximately -67 dBm during intermittent lock from T + 60 seconds until T + 73.5 seconds, when the signal level reached -48 dBm. The airborne received signal strength reached a maximum signal level of -11 dBm at T + 96 seconds, and then gradually decayed throughout the flight to -32 dBm at Atlas-Agena separation, and to -76.5 dBm at T + 409.5 seconds, when lock became intermittent.

Pulses were missing between T + 101.8 and T + 102.9 seconds and between T + 221.8 and T + 124.3 seconds due to momentary loss of signal. This loss is expected when the look angles from the vehicle antenna to the ground station coincide with the nulls in the vehicleborne antenna radiation pattern. A momentary loss of signal also occurred during the booster staging sequence as a result of signal attenuation caused by the Atlas sustainer exhaust plume.

The magnetron current monitor indicated intermittent pulse beacon response until T + 88.5 seconds, as a result of the acquisition problem. The magnetron current monitor indicated good beacon response from T + 88.5 to T + 410.5 seconds, and intermittent beacon response until the magnetron current dropped to zero at T + 426.5 seconds.

The two AGC monitors on the rate beacon indicated that the received signal strengths varied about the threshold level from T + 49.5 until T + 87 seconds. From T + 87 seconds until approximately T + 391.5 seconds the signal levels were greater than -75 dBm. The signal strengths decreased to the threshold sensitivity of the receiver, -82 dBm, at approximately T + 393.5 seconds. The rate beacon phase detector and power output monitors indicated that the received signals were processed and that the return signal was transmitted back to the ground station during the period from T + 87 to T + 403 seconds.

The steering and discrete commands transmitted from the ground station were received and processed by the vehicleborne decoder. Spurious pitch and yaw commands were observed prior to T + 80 seconds, but these commands were inhibited by the airborne flight control system.

Guidance steering was enabled within the airborne flight control at T + 80 seconds, at which time spurious steering commands from the decoder were approximately +10 to -8 percent of maximum command (100 percent equals 2 deg/sec gyro torque rate). The booster steering resulting from these spurious commands was negligible.

No steering commands were transmitted to the vehicle during the period programmed for Mod III guidance commands between T + 100 and T + 110 seconds because the pitch attitude error was less than the computed flight threshold for steering commands. The computed flight threshold is a combination of a predetermined pitch attitude increment plus an uncertainty increment from the estimation of the "true" attitude. The magnitude of the uncertainty increment is dependent upon the amount of tracking data supplied to the computer. The uncertainty increment at T + 100 seconds for normal acquisition, and continuous tracking data, is small. Therefore, the computed flight threshold is approximately equal to the predetermined pitch attitude increment, which, for this mission, was a 1-sigma pitch attitude dispersion.

Because of late acquisition on this flight, the uncertainty increment of the computed flight threshold at T + 100 seconds was larger than for a normal acquisition. Therefore, the computed flight threshold was greater. The computed and expected flight thresholds are shown in figure V-8.

Also shown in figure V-8 is the desired booster attitude correction, for the actual trajectory, that would result in the proper altitude rate at booster engine cutoff. Only a portion of the desired booster attitude correction can be made because steering is stopped when the vehicle attitude error is within the threshold deadband.

Sustainer steering commands started at T + 138.4 seconds. The largest steering command outputs from the decoder were a yaw left command of 55 percent of maximum steering at T + 139.5 seconds and a pitch up command of 94 percent at T + 141 seconds (100 percent equals 2 deg/sec gyro torque rate). Steering commands from the decoder were reduced to within ±10 percent by T + 164.0 seconds, and remained within ±10 percent until sustainer engine cutoff. The amplitude and duration of steering commands indicated normal steering by radio guidance.

The largest steering command outputs from the decoder during vernier solo phase were a 25 percent yaw right command which was reduced to within ±10 percent in 1.5 seconds, and an 80 percent pitch down command that was reduced to within ±10 percent in 3.0 seconds. These commands were within the acceptable limits.

Table V-VI presents the times and durations of the discretes generated by the guidance computer.

TABLE V-V. - ATLAS PITCH PROGRAM, MARINER VENUS 67

Time interval, sec		Step level, deg/sec	
Programmed	Actual	Programmed	Actual
0 to 15	0 to 15	0	0
15 to 35	15 to 35.1	1.018	.9828
35 to 45	35.1 to 45.0	.848	.819
45 to 58	45.0 to 58.1	.509	.4914
58 to 70	58.1 to 70.2	.678	.6552
70 to 82	70.2 to 82	.806	.819
82 to 91	82 to 91	.678	.6552
91 to 105	91 to 105.1	.551	.5733
105 to 120	105.1 to 120	.382	.3276
120 to 128.4	120 to 128.4	.254	.2457
128.4 to sustainer engine cutoff	128.4 to sustainer engine cutoff	.254	(a)

[a]Cannot be determined.

TABLE V-VI. - MOD III RADIO GUIDANCE COMPUTER DISCRETE TIMES, MARINER VENUS 67

Flight event	Guidance computer generated discrete times, seconds after lift-off	Discrete duration, sec
Booster engine cutoff	128.427	0.497
Start Agena restart timer	281.704	.720
Sustainer engine cutoff	296.607	.817
Start agena timer	308.374	.550
Vernier engine cutoff	317.854	.570
Shroud separation	319.927	.497
Atlas-Agena separation	321.927	To end

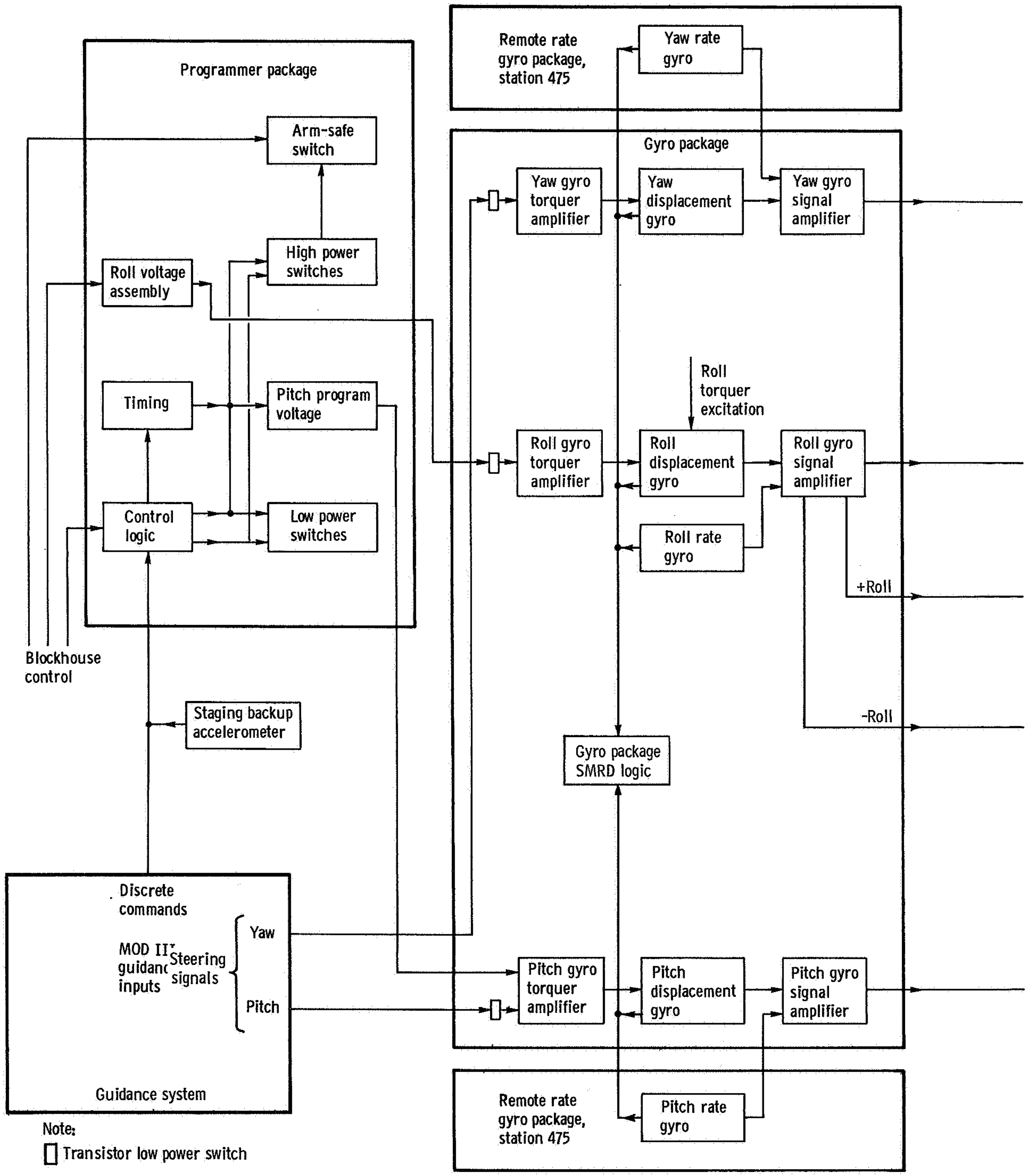
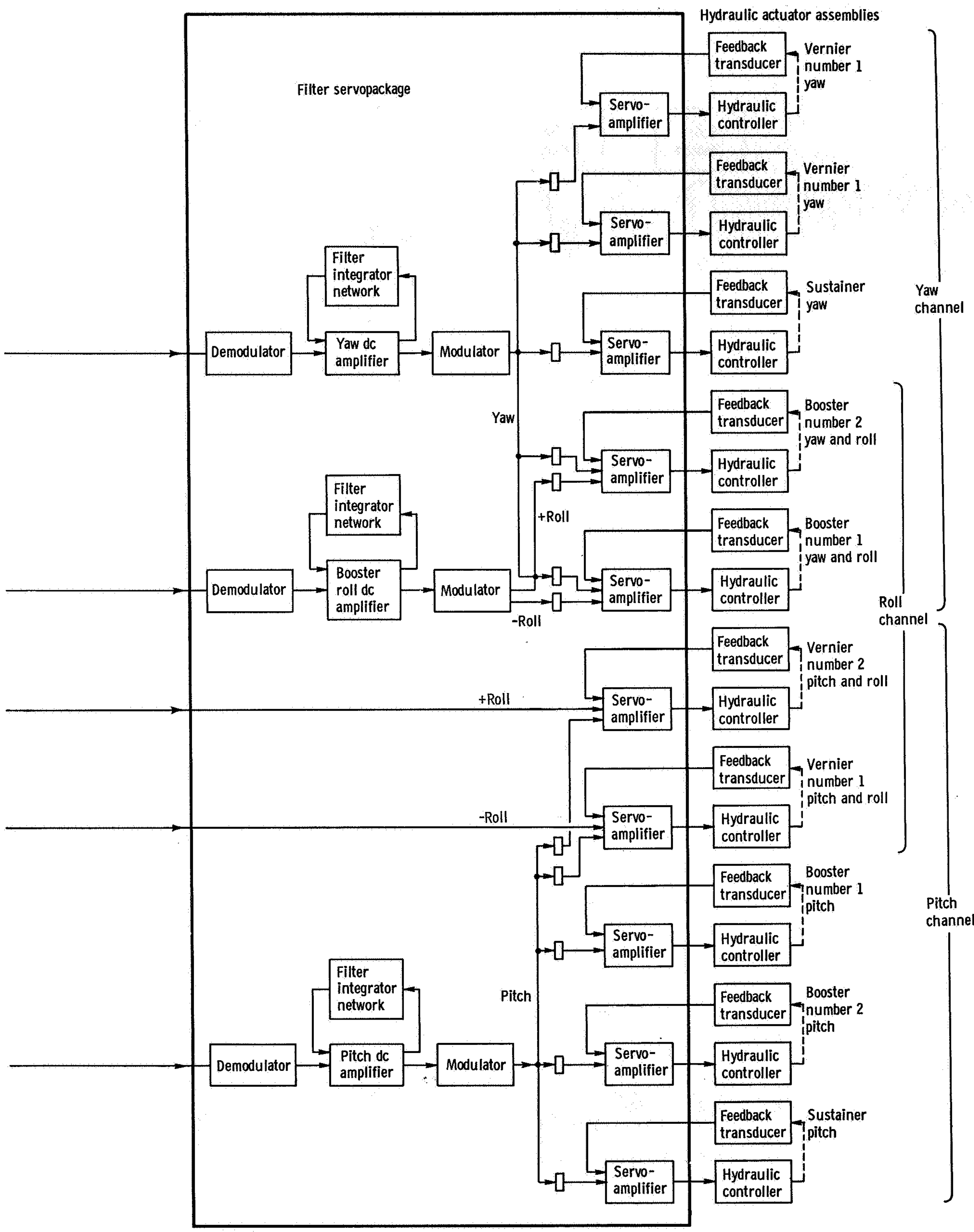


Figure V-6. - Flight control system



block diagram, Mariner Venus 67.

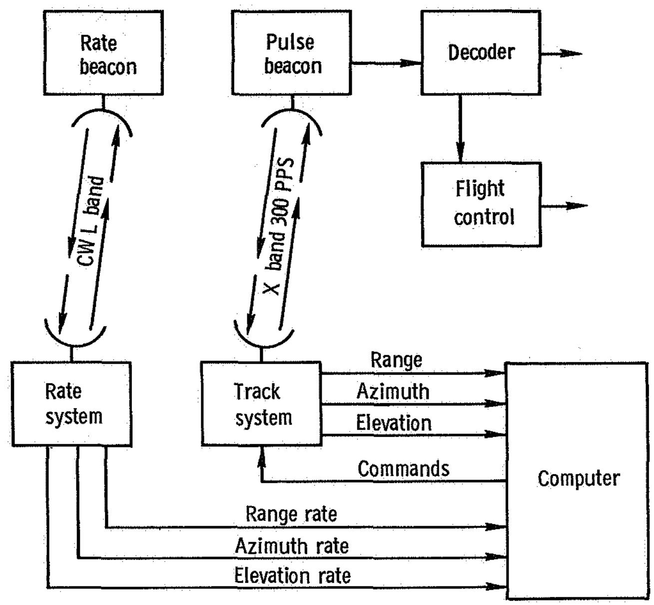


Figure V-7. - MOD III guidance system block diagram.

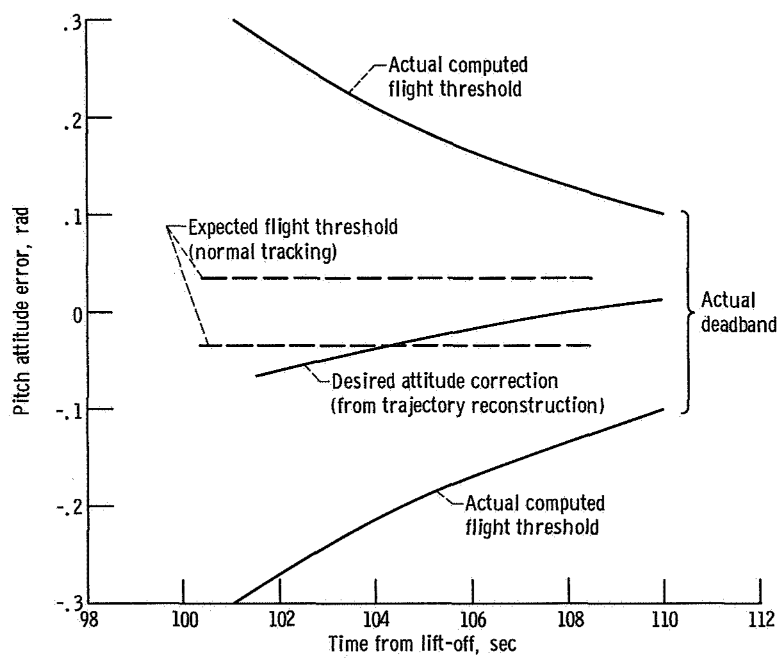


Figure V-8. - Threshold for booster steering, Mariner Venus 67.

ELECTRICAL SYSTEM

by Clifford H. Arth

Description

The Atlas electrical system supplies and distributes power to user systems. The electrical system consists of four 28-volt dc manually activated batteries; a 115-volt ac, three-phase, 400-hertz inverter; a power changeover switch; a distribution box; two junction boxes; and related electrical harnesses. The main 28-volt dc battery supplies power to the flight control system, the airborne radio guidance system, the propellant utilization system, the propulsion system, and the inverter. Another 28-volt dc battery supplies power to the telemetry system, and the remaining two 28-volt dc batteries supply power to the flight termination system. The inverter supplies power to the flight control, the propellant utilization system, and the airborne radio guidance system. Phase A of the inverter is used as a phase reference in the flight control and the radio guidance system.

The vehicle flight control, propulsion, airborne radio guidance. and propellant utilization systems operate from ground regulated dc and ac power sources until 2 minutes prior to lift-off. At this time, the power changeover switch is used to transfer power from ground sources to vehicle electrical power supply.

Performance

Measured electrical system parameters were within specifications throughout flight. Electrical system performance data are shown in table V-VII for selected times.

TABLE V-VII. - ELECTRICAL SYSTEM PERFORMANCE, MARINER VENUS 67

Performance parameter	Units	Specification	Flight values at -				
			Lift-off	Booster engine cutoff	Sustainer engine cutoff	Vernier engine cutoff	Atlas-Agena separation
Main battery voltage, dc	V	$28\begin{cases}+2.0\\-1.5\end{cases}$	27.9	27.8	27.9	28.0	28.1
Inverter frequency	Hz	400±6	399.2	399.2	399.5	400.1	400.1
Phase A voltage, ac	V	115±0.5	115.5	115.5	115.5	115.2	115.0
Phase B voltage, ac	V	115±1.7	116.2	116.0	115.9	115.9	115.8
Phase C voltage, ac	V	115±1.7	115.9	115.9	115.8	115.5	115.5

TELEMETRY SYSTEM

by Edwin S. Jeris

Description

The Atlas telemetry system consists of a telemetry package, a manually activated 28-volt dc battery, associated transducers, wiring harness, and two antennas. Figure V-9 shows the telemetry system. Appendix B summarizes the launch vehicle instrumentation.

The 18-channel Pulse Amplitude Modulation/Frequency Modulation/Frequency Modulation (PAM/FM/FM) telemetry system consists of a transmitter, commutator assemblies, signal conditioning components, and subcarrier oscillators. The telemetry transmitter has an output power level of 3.5 to 6 watts and requires 28 volts dc for operation. The telemetry system is designed to use standard Interrange Instrumentation Group (IRIG) subcarrier channels 1 to 18. Channels 2, 3, 4, and 12 are continuous direct, and the transducer provides the modulating frequency. Channels 5 to 10 are continuous, using subcarrier oscillators. Channel 11 is commutated at 2.5 rps; channel 13 is commutated at 5 rps; channels 15 and 16 are commutated at 10 rps; and channel 18 is commutated at 30 rps. Channels 1, 14, and 17 were not used for this flight. The outputs of all subcarrier channels are multiplexed to allow continuous frequency modulation of the 249.9-megahertz carrier wave.

Performance

Atlas telemetry performance was satisfactory on Mariner Venus 67. One-hundred and sixteen measurements (table V-VIII) were made, and all yielded usable data. No telemetry problems occurred during the countdown or during the flight. Signal strength was adequate during flight except for the expected 1-second loss of signal at booster engine section jettison. This loss of signal occurs as a result of signal attenuation by the Atlas sustainer exhaust plume. Carrier frequency and commutator speeds were stable, and no data playback difficulties were encountered.

The location of the telemetry stations and the telemetry coverage provided are shown in appendix C.

TABLE V-VIII. - ATLAS TELEMETRY

[Blocked-in measurements are either inactive or will indicate less than 5 percent of

Analysis category		Interrange Instrumentation Group									
		1	2	3	4	5	6	7	8	9	10
		Continuous									
Airframe	Adapter										
	Structural							M79A			
	Engine area										
Propulsion system	Engine controls								P77X P347X		
	Propellant feed			P349B	P84B						
	Engine performance										
Control systems	Flight control					S54B M32X S359X	S209V	S254D M30X	S53R	S52R S236X S241X S245X S248X	
	Mod III ground guidance		G364X			Y41X	G363X				
Support systems	Propellant utilization										
	Pneumatic										
	Hydraulic										
	Electrical										E151V
	Range safety					D1V					
Number of measurements		0	1	1	1	5	2	3	3	5	1
Total number of											

information bandwidth prior to launch. They become active either at engine ignition or later in flight.]

(IRIG) channel number and commutation rate								Alternate measurements indicative of performance
11	12	13	14	15	16	17	18	
2.5 rps	Con-tinuous	5 rps	Open	10 rps		Open	30 rps	
Y45T		Y44P						
	U101A							F1P P116P F3P
A743T P15T A745T P16T P61T								
		P26P P344P			P616X			F291P S241X H140P S245X S236X
P117T P530T		P2P P47P P27P P49P P30P P55P		P83B P100P P330P			P1P P56P	
							P6P P59P P28P P60P P29P P339P	P330P U101A
				S61D S62D S63D S190V S384X	S252D S259D S253D S260D S255D S261D S256D S290X S257D S291X S258D			M79A P77X P347X P616X U101A
		G296V G298V G354V		G4C G280V G82E G282V G3V G287V G279V G288V	G590V G592V G591V G593V			S61D S236X S62D S241X S63D S243X S52R S248X S53R U101A S54R
		P529D U113V P830D U132V		U112V	U80P U81P		U134V U605V U135V	
P247T		F125P F288P F246P F291P		F1P F3P			F116P	
		H3P		H33P H140P H130P H224P			H601P	
		E95V E96V					E28V E52V E31V E53V	
				D1V D7V	D3X			
9	1	23	0	25	19	0	17	

measurements - 116

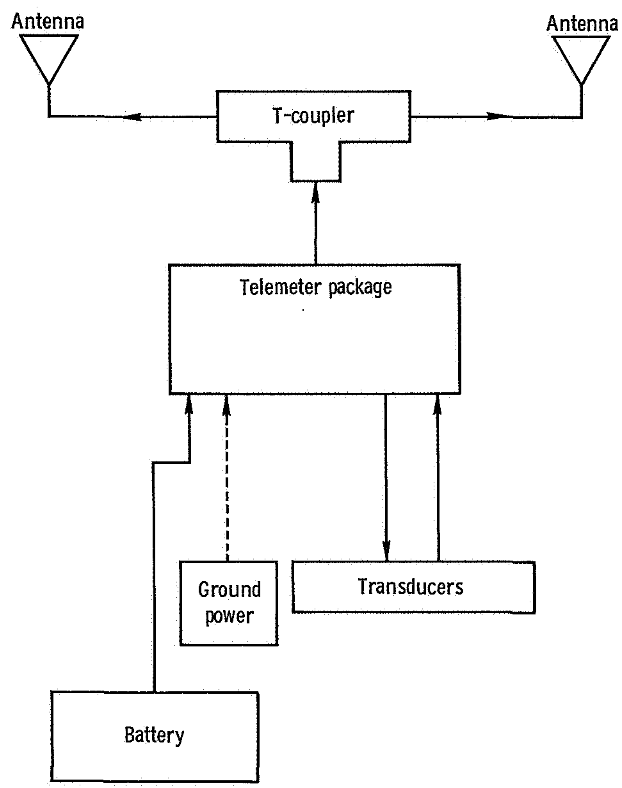


Figure V-9. - Atlas airborne telemetry system, Mariner Venus 67.

FLIGHT TERMINATION SYSTEM

by Edwin S. Jeris

Description

The Atlas contains a vehicleborne flight termination system (fig. V-10) which is designed to function on receipt of command signals from the ground stations. This system includes two receivers and two batteries for redundancy, a power control unit, an electrical arming unit, and a destructor. The two batteries operate independently of the vehicle main power system.

The Atlas flight termination system provides a highly reliable means of shutting down the engines only, or shutting down the engines and destroying the vehicle. When the vehicle is destroyed in the event of a flight malfunction, the tank is ruptured with a conical-shaped charge, and the liquid propellants are dispersed. The operation of the flight termination system is under command of the Range Safety Officer only.

Performance

Performance of the flight termination system was satisfactory. Prelaunch checks were completed without incident. The only measurements telemetered were receiver 1 automatic gain control, engine cutoff, and destruct commands. The automatic gain control measurement on receiver 1 indicated that the capability to terminate flight was maintained throughout powered flight. Minimum signal strength measured at the receiver on Mariner Venus 67 was 47 microvolts except for the expected loss of signal at booster engine section jettison. Five microvolts is the minimum required signal strength for receiver operation. Receiver 2 was not instrumented. No flight termination commands were required, nor were any commands inadvertently generated by any vehicle system.

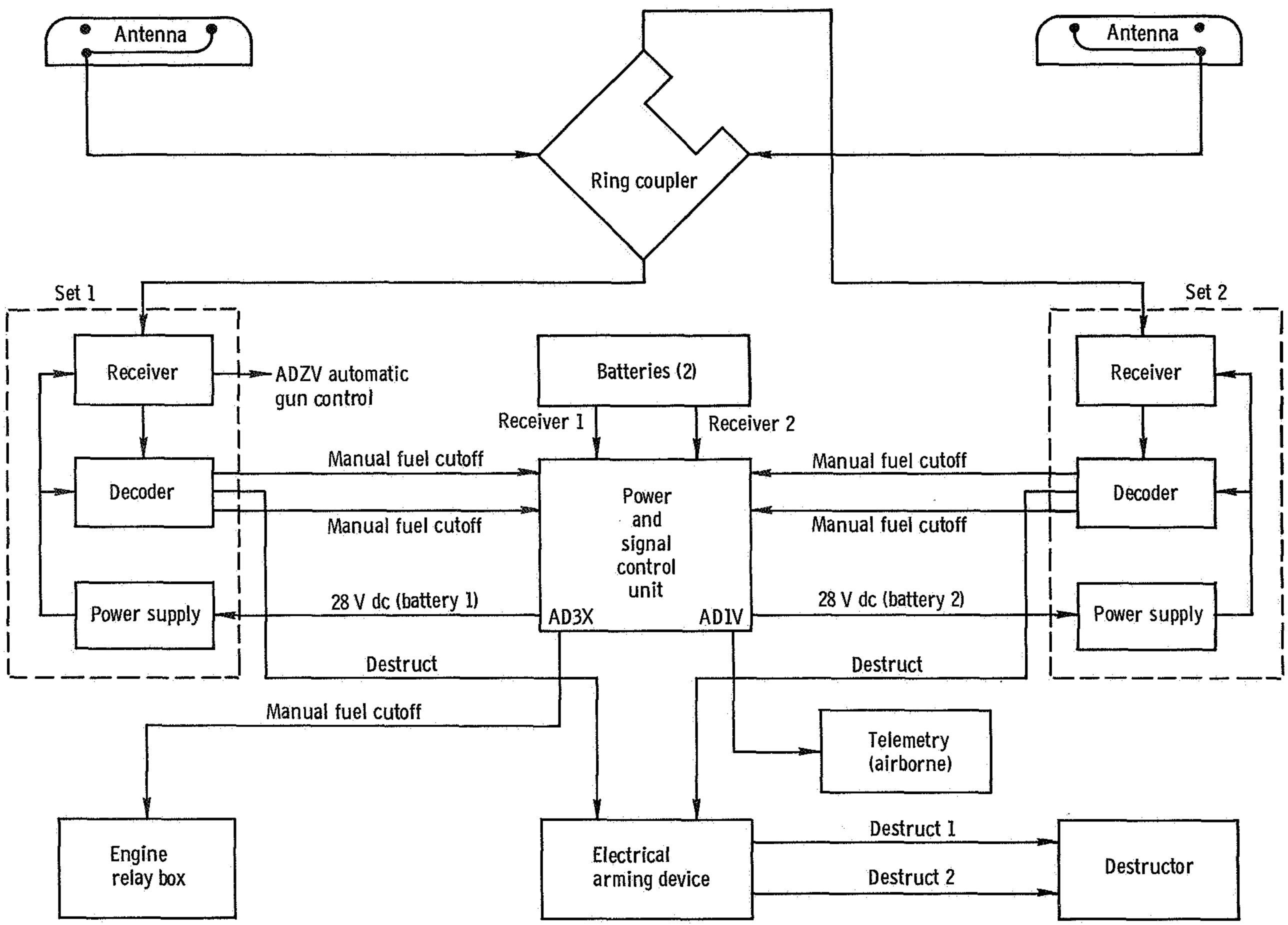


Figure V-10. - Atlas flight termination system, Mariner Venus 67.

VI. AGENA VEHICLE SYSTEM PERFORMANCE

VEHICLE STRUCTURE SYSTEM

by Robert N. Reinberger

Description

The Agena vehicle structure system (fig. VI-1) consists of four major sections: forward section, tank section, aft section, and booster adapter assembly. Together they provide the aerodynamic shape, structural support, and environmental protection for the vehicle. The forward section is basically an aluminum structure with beryllium and magnesium panels. This section encloses most of the electrical, guidance, and communication equipment and provides the mechanical and electrical interface for the spacecraft adapter and shroud. The tank section consists of two integral aluminum propellant tanks, with a sump below each tank to assure the supply of propellants for engine starts in space. The aft section consists of an engine mounting cone structure and an equipment mounting rack. The magnesium alloy booster adapter assembly consists of the basic adapter section and the Atlas adapter extension. This assembly supports the Agena and remains with the Atlas after Atlas-Agena separation.

Performance

The measured dynamic environment of the structure system was within design limitations. The data are presented in appendix D.

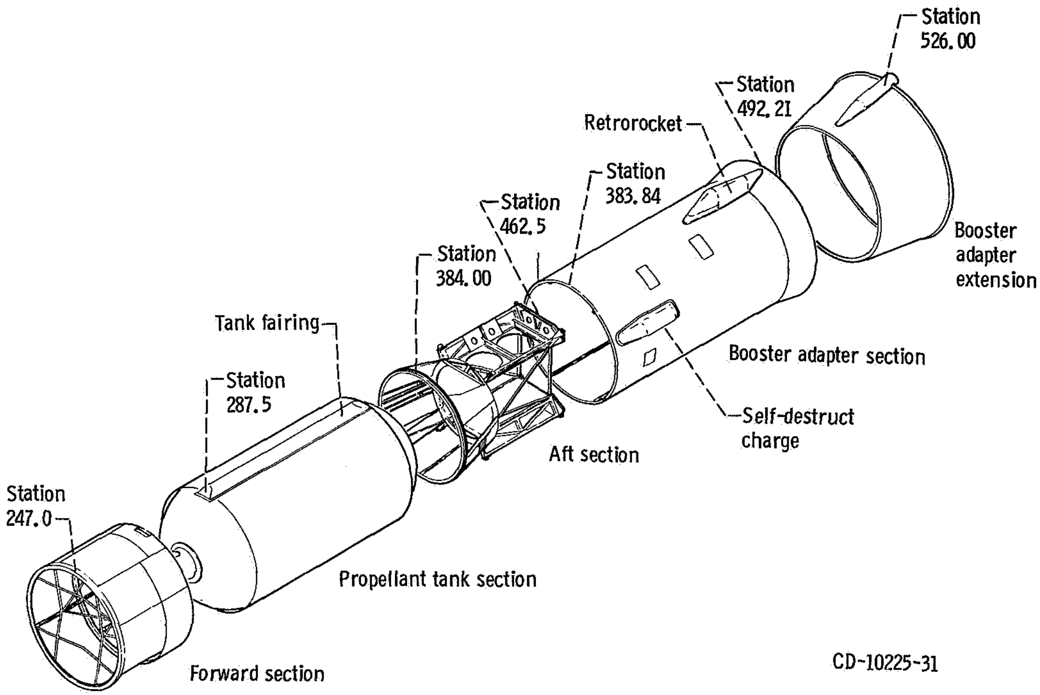


Figure VI-1. - Agena vehicle structure system, Mariner Venus 67.

SHROUD-ADAPTER SYSTEM

by C. Robert Finkelstein

Description

The shroud-adapter system, shown in figures VI-2 and VI-3, consists of an all-metal, "over-the-nose" shroud and a spacecraft adapter. The shroud provides environmental protection for the spacecraft prior to and during launch. The adapter is the transition section which attaches both the shroud and the spacecraft to the Agena.

The shroud is 3.98 meters (13.0 ft) long and weighs 165.8 kilograms (363 lb). It consists of a 1.66-meter (5.43-ft) diameter cylindrical section, a 15° half-angle conical section, an ogive section, and a 0.72-meter (2.4-ft) diameter hemispherical nose cap. The nose cap is made of beryllium; all other sections are made of magnesium alloy. The shroud is strengthened by magnesium alloy internal rings to which polished aluminum liners are attached to protect the spacecraft from thermal radiation. The shroud is clamped to the spacecraft adapter by a V-band which is tensioned to approximately 30 000 newtons (6750 lb). A new, spacecraft electrical umbilical door was used on this shroud. This door is held closed during launch by a spring and linkage arrangement.

The spacecraft adapter is approximately 1.52 meters (5 ft) in diameter and is 0.21 meter (0.70 ft) high. It is made of aluminum and magnesium and is bolted to the forward end of the Agena. A magnesium diaphragm attached to the adapter isolates the shroud cavity from the Agena. During ascent, a valve in this diaphragm vents the shroud cavity through the Agena. The spacecraft is clamped to the adapter by a V-band which is tensioned to 11 125±334 newtons (2500±75 lb).

Shroud jettison is initiated by a radio guidance discrete approximately 2 seconds after Atlas vernier engine cutoff. At this time, Agena electrical power is used to fire two pyrotechnically-actuated release devices in the shroud separation V-band. The firing of either release device will effect shroud separation. When the V-band is released, four pairs of spring loaded pushrods, which are in the shroud and thrust against the spacecraft adapter, provide the energy to eject the shroud over the nose of the spacecraft at a relative velocity of 2 meters per second (7 ft/sec).

The temperature in the shroud is controlled; cold air is circulated through an external blanket to maintain an acceptable shroud cavity temperature. This blanket is automatically removed from the shroud at lift-off. The humidity in the shroud is also controlled. A nitrogen purge system introduces dry nitrogen at a flow rate of approximately 2.26 cubic meters per hour (80 cu ft/hr) into the shroud cavity near the top, and nitrogen exhausts through the spacecraft adapter. This purge system is manually disconnected prior to gantry rollback.

The shroud-adapter system was instrumented with one accelerometer mounted in the adapter, two shroud separation switches, two semiconductor strain gages to measure tension in the spacecraft separation V-band, a temperature transducer attached to the diaphragm, and a pressure transducer mounted in the adapter.

The Mariner spacecraft and the shroud are mated to the adapter in an environmental clean room, and the complete assembly is then transported to the launch pad and mated to the Agena. Since the spacecraft carries fuel, the clean room was also required to be an explosive-safe area.

Performance

During ascent, the shroud compartment pressure decayed in the expected manner, indicating proper valve operation.

The temperature measured at the adapter diaphragm remained at a satisfactory level throughout the ascent part of the flight. The temperature was 290 K (62° F) at lift-off, decreased to 283.9 K (51° F) about 100 seconds after lift-off, and increased to 287.8 K (58° F) at the time of shroud jettison.

Shroud V-band pyrotechnics were fired at T + 320.1 seconds and both separation switches closed, which indicated that shroud jettison was properly initiated. Neither the spacecraft axial accelerometer nor the adapter axial accelerometer showed any unusual disturbance during shroud jettison, indicating that the shroud did not touch the spacecraft and was properly ejected. The Agena vehicle was stable at this time, and no measurable rates in roll, pitch, or yaw developed as a result of shroud jettison.

Spacecraft V-band pyrotechnics were fired at T + 1576.7 seconds. The tension values in both instrumented turnbuckles in the V-band fell to zero tension simultaneously, which indicated that spacecraft separation was normal and satisfactory. The velocity of the spacecraft relative to the Agena was 0.81 meter per second (2.66 ft/sec), which was the predicted value.

During final assembly, the spacecraft V-band was tensioned to 11 125±334 newtons (2500±75 lb). At this time, the actual tension values measured by the two instrumented turnbuckles in the V-band were 11 392 and 11 236 newtons (2560 and 2525 lb). At spacecraft separation, the tension values were 11 040 and 11 984 newtons (2481 and 2693 lb).

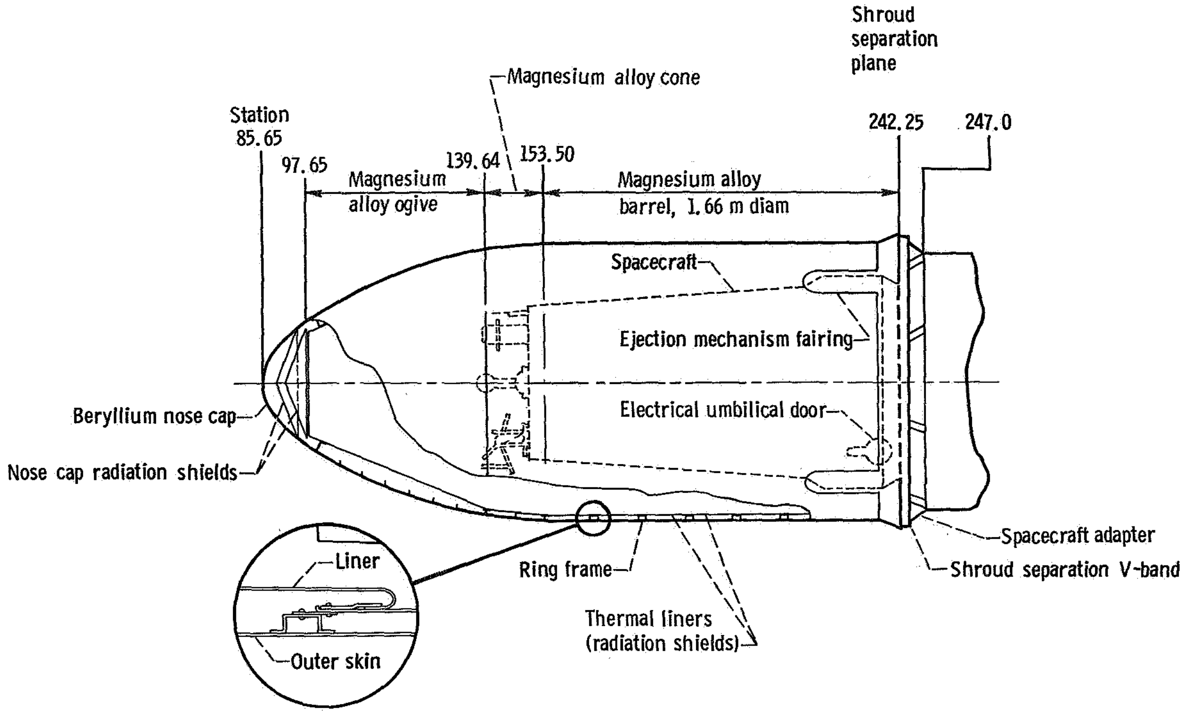


Figure VI-2. - Shroud system, Mariner Venus 67.

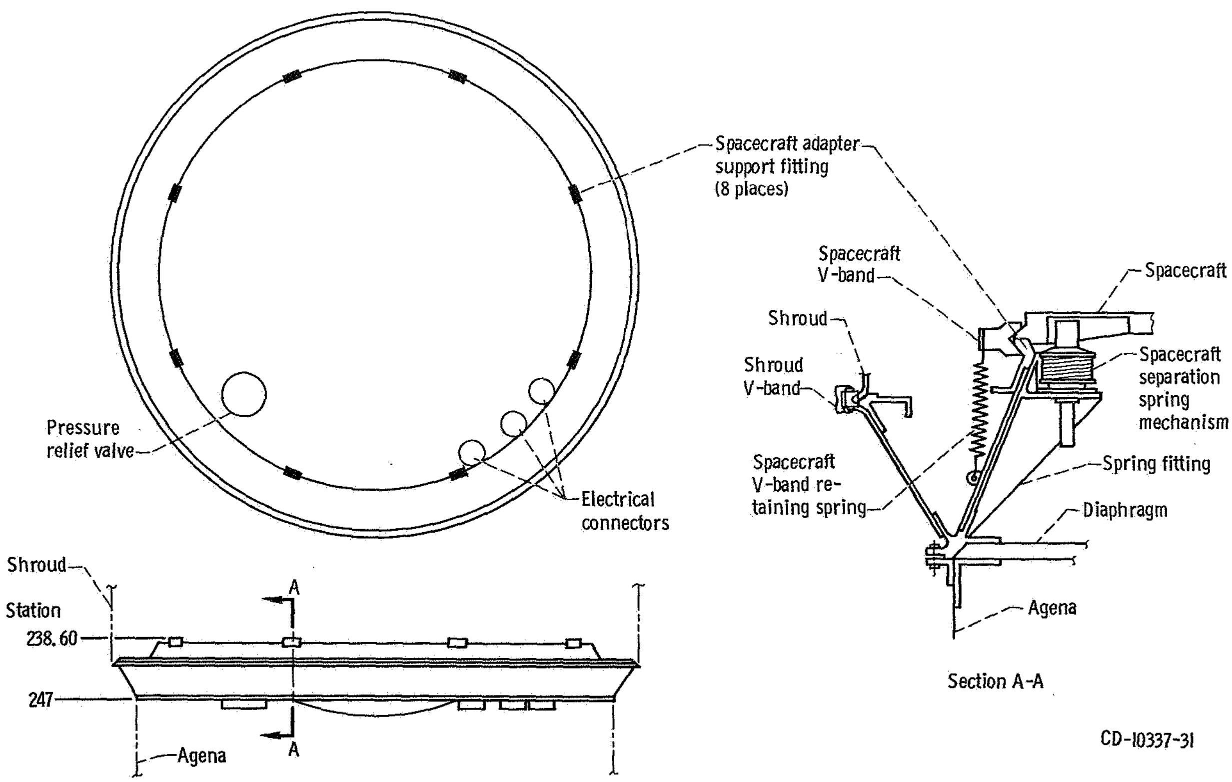


Figure VI-3. - Spacecraft adapter, Mariner Venus 67.

PROPULSION SYSTEM

by Robert J. Schroeder

Description

The Agena propulsion system, as shown in figure VI-4, consists of a propellant tank pressurization system, a propellant management system, and an engine system. Also included under the propulsion system are the Atlas-Agena separation system, an Agena posigrade rocket, and vehicle pyrotechnic devices.

The propellant tank pressurization system consists of a helium supply tank and a pyrotechnically operated helium control valve to provide the required propellant tank pressures. Before lift-off, the ullage volume in the propellant tanks is pressurized with helium from a ground supply source. The helium control valve is activated after Agena engine first ignition to permit helium gas to flow from the supply tank through fixed-area flow orifices to each propellant tank. After Agena engine first cutoff, the helium control valve is again activated to isolate the oxidizer tank from the helium supply. This prevents the mixing of oxidizer and fuel vapors that could occur when the pressure in the helium tank decreases to a level equal to the pressure in the propellant tanks. Since the helium supply is essentially depleted at the time of Agena engine first cutoff; pressurization for the oxidizer tank and the fuel tank, during the Agena engine second burn, is provided by the residual helium pressure in each propellant tank.

The propellant management system consists of the following major items: propellant fill disconnects to permit the loading of fuel and oxidizer, feed lines from the propellant tanks to the engine pumps, tank sumps to retain a sufficient amount of propellants for engine restart after a zero-gravity coast, and an electric motor driven propellant isolation valve in each feed line. The propellant isolation valves are normally open at the time of lift-off, closed after the end of the Agena engine first burn, and opened 2 seconds before starting the Agena engine second burn. When closed, these valves isolate propellants in the tanks from the engine pump inlets and provide an overboard vent for propellants trapped in the engine pumps.

The Agena engine is a Bell Aerosystems Company Model 8096 liquid bipropellant engine which uses unsymmetrical dimethylhydrazine (UDMH) fuel and inhibited red fuming nitric acid (IRFNA) oxidizer. Rated thrust in a vacuum is 71 171 newtons (16 000 lb) with a nozzle expansion area ratio of 45:1. The engine has a regeneratively cooled thrust chamber and a turbopump-fed propellant flow system. Turbine rotation is initiated for each engine firing by a solid-propellant starter charge. The turbine is driven during steady-state operation by hot gas produced in a gas generator. Propellants to the gas generator are supplied by the turbopump. An oxidizer fast-shutdown system consisting of a pyrotechnically operated valve and a high-pressure nitrogen storage cylinder is used to rapidly close the main oxidizer valve at first-burn cutoff. Engine thrust vector control is provided by the gimbal-mounted thrust chamber. A pair of hydraulic actuators provide the force for thrust chamber movement in response to signals produced by the Agena guidance system.

Atlas-Agena separation is accomplished by firing a Mild Detonating Fuse which cuts the booster adapter circumferentially near the forward end. The Atlas and booster adapter are then separated from the Agena by firing two solid-propellant retrorockets mounted on the booster adapter. Rated average sea-level thrust of each retrorocket is 2180 newtons (490 lb) with an action time of 0.93 second.

A solid-propellant posigrade rocket is mounted on the aft rack of the Agena with a rated average vacuum thrust of 609 newtons (137 lb) and with an action time of 16.6 seconds. The posigrade rocket is fired after spacecraft separation to bias the Agena trajectory away from Venus.

Pyrotechnic devices are used to perform a number of functions on the Agena. These devices include squibs, igniters, detonators, and explosive bolt cartridges. Squibs are used to open and close the helium control valve, to eject the horizon sensor fairings, to permit repositioning of the horizon sensor heads and to activate the oxidizer fast-shutdown system. Igniters are used for the main engine solid-propellant starter charges, the retrorockets (for Atlas-Agena separation), and the posigrade rocket. Detonators are used for the command destruct charge and the Mild Detonating Fuse separation charge. Explosive bolt cartridges are used to rupture the shroud V-band release devices.

Performance

The Agena engine first burn was initiated by the primary sequence timer at T + 380.4 seconds. Voltage levels on the engine switch group monitor indicated a normal start sequence of the engine control valves. Ninety percent combustion chamber pressure was reached 1.2 seconds later at T + 381.6 seconds. The average steady-state thrust was 71 822 newtons (16 162 lb) compared with an expected value of 72 105 newtons (16 210 lb). The Agena engine first burn was terminated by velocity meter cutoff command at T + 525.3 seconds. The engine burn duration, measured from 90 percent chamber pressure to velocity meter cutoff command was 143.7 seconds. This was 0.4 second longer than the expected value of 143.3 seconds. The actual burn time and thrust level indicate that engine performance was within the allowable 3-sigma limit.

The propellant tank pressurization system supplied the required tank pressures. This was evidenced by the fuel and oxidizer pump inlet pressure values which were within 1.4 N/cm^2 (2 psi) of the expected values during the first burn.

The propellant isolation valves (PIV) were commanded to close at T + 533.4 seconds. The fuel PIV and oxidizer PIV functioned normally as evidenced by the valve position measurement and the decrease in pump inlet pressures.

Near the end of the Agena coast period, the PIV(s) were commanded to open at T + 1317.8 seconds and both PIV(s) functioned normally. The engine ignition sequence occurred 2 seconds later at T + 1319.8 seconds. Voltage levels on the engine switch group monitor indicated a normal start sequence. Ninety percent combustion chamber pressure was reached 1.1 seconds later, indicating a normal start transient. Engine performance appeared normal for the next 5 seconds. At T + 1326 seconds a minor decrease in combustion chamber pressure of approximately 3.1 N/cm^2 (4.5 psi) occurred and lasted for 1.5 seconds. For the next 14 seconds the chamber pressure remained fairly constant at 353.4 N/cm^2 (512.5 psia). This was followed by a gradual decline in chamber pressure until at T + 1354 seconds when a major decrease in chamber pressure commenced. Chamber pressure dropped from 348.2 N/cm^2 (507 psia) to 329.6 N/cm^2 (478 psia). Recovery to 349.6 N/cm^2 (507 psia) was achieved 1.2 seconds later. Chamber pressure continued to rise and leveled off at approximately 362.7 N/cm^2 (526 psia). At T + 1377.5 seconds the chamber pressure began a gradual decrease and at the time of engine shutdown reached 356.5 N/cm^2 (517 psia). The chamber pressure averaged 2.55 percent above the expected level after recovery from the major decrease. A comparison of actual and expected second-burn chamber pressures is shown in figure VI-5. Second burn was terminated by velocity meter cutoff command at T + 1415.35 seconds. The actual thrust duration, measured from 90 percent chamber pressure to velocity meter cutoff command was 94.4 seconds compared with an expected time of 95.59 seconds. The shorter burn time is attributed to the higher than expected chamber pressure experienced during the last 60 seconds of engine burn.

The anomalous behavior of the chamber pressure during the second burn was substantiated by flight data on the turbine speed and both fuel and oxidizer venturi inlet pressures. Despite this engine anomaly, there was no adverse effect on the desired Agena transfer trajectory.

The propellant tank pressures appeared satisfactory during the engine second burn. This was evidenced by the pump inlet pressures which were normal for approximately the first 20 seconds of engine operation. After this time, the pump inlet pressures showed an increasing amount of peak-to-peak fluctuation that was greater than normal. This increase in fluctuation of the pump inlet pressures was due to the anomaly which occurred in the turbopump assembly and was not the result of variation in tank pressures.

The Atlas-Agena separation system performance was normal. Separation was commanded by the ground radio guidance and occurred at T + 322.3 seconds when the Mild Detonating Fuse and the two retrograde rockets were ignited. Complete separation of the Atlas and the Agena was accomplished in 2.0 seconds.

The Agena posigrade rocket was ignited after spacecraft separation at T + 1876.8 seconds. Data from the velocity meter accelerometer indicated a normal action time of the posigrade rocket.

All of the Agena pyrotechnic devices performed satisfactory. Momentary electrical short circuits of some squibs occurred after firing, but this did not produce any adverse effect on vehicle performance. The Agena command destruct charge and detonator were not activated during the flight since the Atlas and Agena flight trajectories were within the allowable flight corridor.

Historical Discussion of Engine Anomaly

The type of engine anomaly experienced on Mariner Venus 67 had previously occurred on several NASA and U.S. Air Force missions. Consequently, an extensive engineering and testing program was established to determine the cause of the anomaly. This program was still in progress at the time of the Mariner Venus 67 launch, but preliminary data were available. These data indicated that the engine anomaly could possibly result from a bearing failure within the turbine pump assembly. On the basis of this information, certain interim changes were made to the turbine pump assembly on the Lunar Orbiter IV vehicle and the Mariner Venus 67 vehicle. The Lunar Orbiter IV flight in May 1967, which preceded the Mariner Venus 67 flight, did not experience the engine anomaly. However, the anomaly occurred on the Mariner Venus 67 flight. The engineering and test program was finally completed about 6 months after the Mariner Venus 67 launch and recommended changes to the turbine pump assembly were finalized. These final changes were subsequently incorporated on the Agena vehicle and to date have apparently corrected the engine anomaly.

The interim changes made to the turbine pump assembly on Lunar Orbiter IV and Mariner Venus 67 vehicles are listed below. Also listed for information are the final changes recommended after completion of the engineering and testing program:

(1) Interim changes

(a) The ball bearings (stainless steel) used on the fuel pump drive shaft and the oxidizer pump drive shaft were replaced with bearings having an internal radial clearance on the loose side of the allowable tolerance.

(b) The fuel pump drive shaft secondary seal (butyl rubber) was removed and replaced with the same type of seal from stock.

(c) A gear case oil flush procedure was used during engine prelaunch servicing to provide additional lubricant to the bearings.

(d) The lubricating oil used in the gear box was restricted to a specific supplier of MIL-L-7808D lubricating oil.

(2) Final recommended changes

(a) A new type ball bearing (high-temperature tool steel) is to be used on the oxidizer pump drive shaft, the fuel pump drive shaft, and the turbine drive shaft.

(b) A new type seal (graphite-filled tetrafluoroethylene) is to be used for the fuel pump drive shaft secondary seal.

(c) Same as interim change (c) for Lunar Orbiter IV and Mariner Venus 67.

(d) Same as interim change (d) for Lunar Orbiter IV and Mariner Venus 67.

(e) A new type seal incorporating a bellows damper is to be used for the turbine hot gas drive seal.

(f) New type gears with improved gear tooth profile and surface finish are to be used in the turbine pump drive assembly.

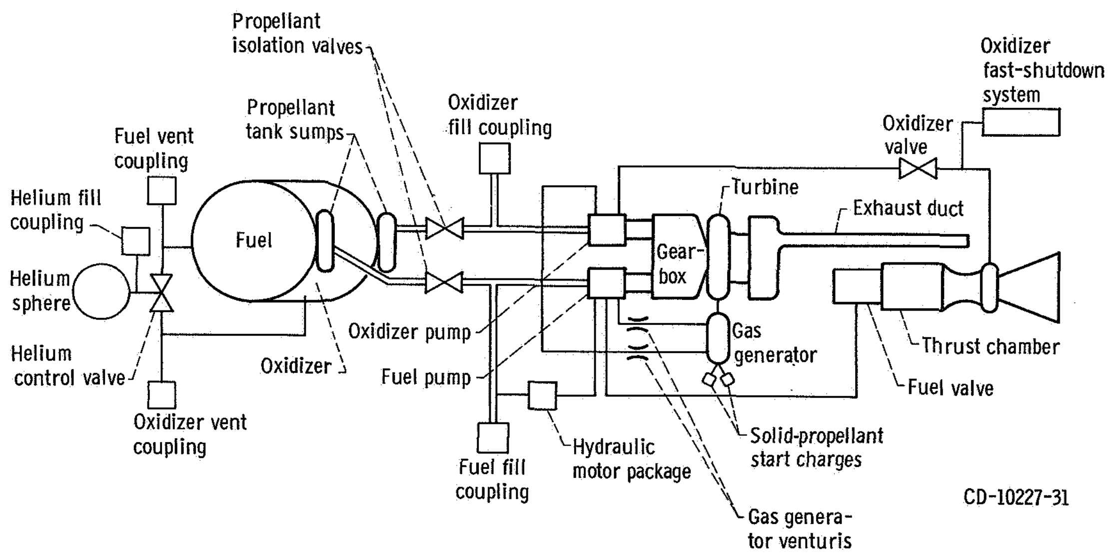


Figure VI-4. - Agena propulsion system schematic, Mariner Venus 67.

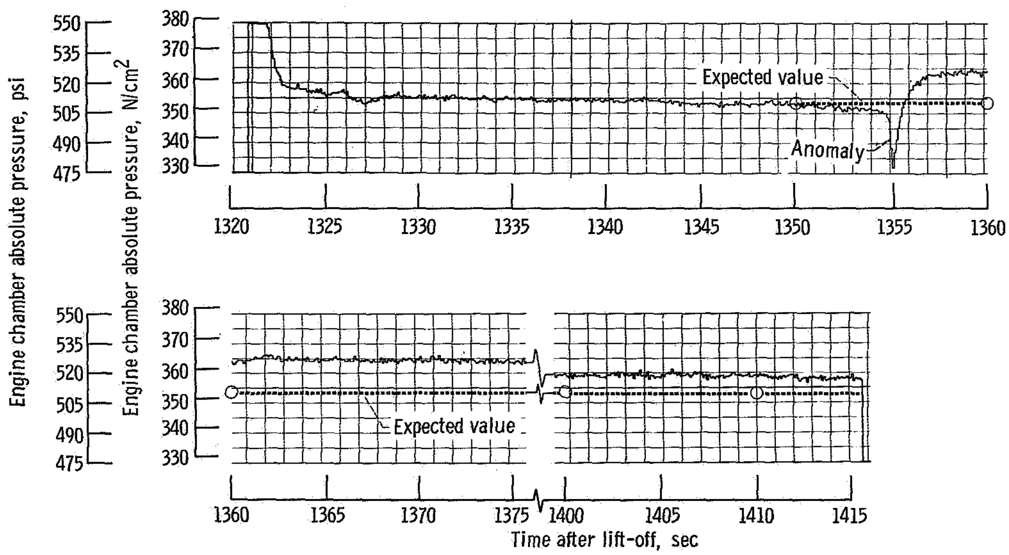


Figure VI-5. - Agena engine chamber pressure during second burn, Mariner Venus 67.

GUIDANCE AND FLIGHT CONTROL SYSTEM

by Howard D. Jackson

Description

The Agena guidance and flight control system performs the vehicle guidance, control, and flight programming functions necessary to accomplish the vehicle mission after Atlas-Agena separation. The system consists of three subsystems: guidance, control, and flight programming. A block diagram of the system is shown in figure VI-6.

The Agena guidance subsystem consists of an inertial reference package (IRP), horizon sensors, velocity meter, and a guidance junction box. Primary attitude reference is provided by three orthogonal rate-integrating gyroscopes in the IRP. The infrared horizon sensors, consisting of a left and right optical sensor (head) provide corrections in pitch and roll to the IRP as required. Yaw attitude is referenced to the attitude of the Atlas at the time of Atlas-Agena separation and is then corrected by gyrocompassing techniques during long (greater than 10 min) coast periods. Longitudinal acceleration is sensed by the velocity meter accelerometer, and the velocity meter counts down the first-burn "velocity-to-be-gained" binary number. The velocity meter counter then generates a signal to terminate engine thrust when the vehicle velocity increases by a predetermined increment. The second-burn velocity-to-be-gained number is transferred from a storage register to the counting register after Agena engine first cutoff.

The Agena flight control subsystem, which controls vehicle attitude, consists of a flight control electronics unit, a cold gas attitude system, a hydraulic control system, and a flight control junction box. Attitude error signals from the IRP are conditioned and amplified by the flight control electronics to operate the cold gas and hydraulic systems. During Agena coast periods, the cold gas system consisting of six thrusters provides roll, pitch, and yaw control. These thrusters operate on a mixture of nitrogen and tetrafluoromethane. During powered flight the hydraulic system provides pitch and yaw control by means of two hydraulic actuators which gimbal the Agena engine thrust chamber; roll control is provided by the cold gas system. A patch panel in the flight control junction box provides the means for varying the interconnections of the guidance and flight control system to suit mission requirements.

The flight programming subsystem uses sequence timers to program Agena flight events. A sequence timer provides 22 usable, discrete event times with multiple switch closure capability and has a maximum running time of 6000 seconds. Two timers (a primary and restart timer) are used for lunar or interplanetary probe missions. The primary sequence timer is started by a ground-initiated radio guidance command. The start time is determined by the ground-based computer after compensating for the trajectory dispersions of the booster. These commands are received by the Atlas and decoded, and a signal is then sent to the Agena. The primary sequence timer is programmed to stop during the Agena coast period. Prior to Agena second ignition the primary timer is restarted by the restart sequence timer. The restart time controls the duration of the coast period so that the spacecraft will be at the proper injection point after second burn. The restart time is established by the following means:

(1) Coarse adjustments are made on the restart sequence timer for various launch days.

(2) Time is run off the restart sequence timer in 45-second increments during the countdown.

(3) A radio guidance command, which starts the restart sequence timer in flight, provides fine adjustments to compensate for lift-off time and trajectory.

Performance

The guidance and flight control system performance was satisfactory throughout flight. All flight events were initiated within tolerance by the sequence timers. A comparison of the expected and actual times of programmed events is given in appendix A. The rates imparted to the Agena at Atlas-Agena separation and the attitude errors that existed following separation were within the range of values experienced on previous flights and are shown below:

Rates imparted to Agena at separation, deg/sec			Attitude errors at cold gas activation, deg		
Yaw	Roll	Pitch	Yaw	Roll	Pitch
0.43 Left	0.13 CCW[a]	(b)	0.8 Left	0.1 CCW[a]	0.2 Up

[a]Clockwise (CW) and counterclockwise (CCW) roll reference applies when looking forward along the Agena longitudinal axis (see fig. VI-7).
[b]Negligible.

The cold gas attitude control system reduced these errors to within the dead band limits of $\pm 0.2^{\circ}$ pitch, $\pm 0.18^{\circ}$ yaw, and $\pm 0.6^{\circ}$ roll in 3.5 seconds. The vehicle then completed a programmed pitchdown of 10°, and the programmed geocentric rate of 3.21 degrees per minute pitchdown was applied. For the Agena first burn, the pitch horizon sensors were set at a pitch bias angle of $+4.18^{\circ}$ (nose up), and the vehicle was stable in all axes by the time of first ignition.

Gas generator spinup at Agena engine first ignition resulted in a roll rate and induced a maximum displacement error as follows:

Roll rate, deg/sec	Maximum displacement error, deg	Time to reverse initial rate, sec
0.2 CW	2.8 CW	1.2

Minimal attitude control was required during the Agena engine first burn. Engine shutdown was commanded by the velocity meter after the vehicle had attained the required velocity increment.

The roll transients, caused by engine shutdown (i.e., turbine spin and turbine exhaust decay) were as experienced on similar previous flights. The time required to reduce the roll excursions to within the cold gas attitude control dead bands was 14 seconds. Approximately 20 seconds after Agena engine shutdown, the programmed geocentric pitch rate was increased to 4.20 degrees per minute pitchdown, and the horizon sensor bias angle was decreased to 0°. The horizon sensor left head was disabled during the coast period to preclude sun interference. Accomplishment of the disable was verified preceding second-burn ignition since no telemetry data is available for the initiation of the event. The cold gas attitude control system was transferred to ascent mode during the time of left head disable to minimize attitude error cross coupling. Horizon sensor and cold gas attitude control data show that the vehicle maintained the proper attitude in the coast phase.

Gas generator turbine spinup at Agena engine second ignition resulted in a roll rate and induced a maximum displacement error as follows:

Roll rate, deg/sec	Maximum displacement error, deg	Time to reverse initial rate, sec
1.8 CW	2.6 CW	1.1

Minimal attitude control was required during the Agena engine burn. Engine shutdown was commanded by the velocity meter after the vehicle had attained the required velocity increment.

Velocity meter accelerometer data during the period of second burn displayed intermittent dc voltage level shifts of approximately ±15 percent of full scale. These shifts are the result of double pulsing of the velocity meter accelerometer which occurs when vehicle acceleration levels exceed the limit for the single pulse mode of operation. This transition in pulse rate normally occurs at 6.5±1.5 g's.

Subsequent to spacecraft separation, the left horizon sensor head was re-enabled and a posigrade maneuver was successfully accomplished. This posigrade maneuver consisted of the vehicle yawing 27° to the right and the firing of the posigrade rockets.

The Agena attitude control gas usage was less than anticipated. A comparison of predicted and actual gas usage is given in table VI-I. These data indicate that the Agena attitude was stable throughout flight.

TABLE VI-I. - ATTITUDE CONTROL GAS LOADING REQUIREMENTS

AND USAGE, MARINER VENUS 67

Flight sequence	Units	Predicted nominal	Actual
Lift-off	kg	13.5 load	13.8 loaded
	lb	29.7	30.5
Agena first coast	kg	2.2 usage	1.2 used
	lb	4.8	2.6
Agena engine first burn	kg	1.3 usage	0.86 used
	lb	2.8	1.9
Parking orbit coast	kg	0.36 usage	0.18 used
	lb	0.8	0.4
Agena engine second burn	kg	1.0 usage	0.6 used
	lb	2.2	1.4
Spacecraft separation and postseparation maneuvers	kg	1.4 usage	1.4 used
	lb	3.0	3.0
Remaining at loss of signal	kg	7.3	9.6
	lb	16.1	21.2

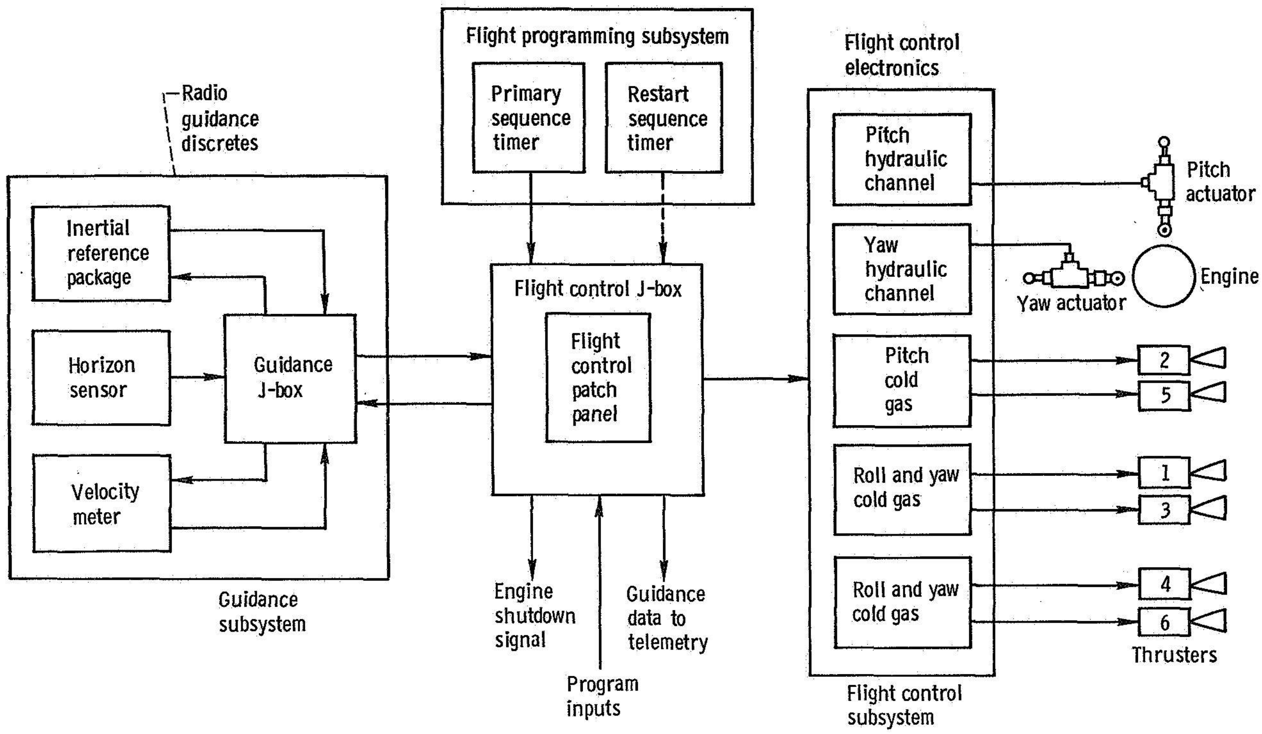


Figure VI-6. - Agena guidance and flight control system: block diagram, Mariner Venus 67.

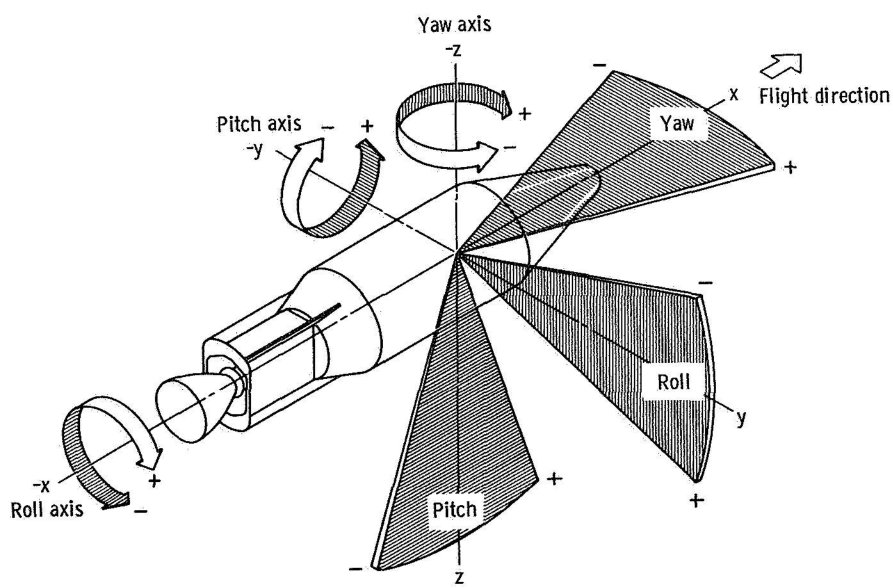


Figure VI-7. - Agena vehicle axes and vehicle movement designations, Mariner Venus 67.

ELECTRICAL SYSTEM

by Edwin R. Procasky

Description

The Agena electrical system (fig. VI-8) supplies all power, frequency, and voltage requirements for the guidance, telemetry, flight termination, propulsion, and pyrotechnic systems. The electrical system consists of two silver-zinc primary batteries, two nickel-cadmium secondary batteries, two dc-dc converters, and one solid-state three-phase inverter and associated electrical harnesses.

The two primary batteries, with a minimum rating of 405 watt-hours each, provide power to all systems except the flight termination system, which is powered by the two secondary batteries. The three-phase inverter supplies regulated 115 volts ac (rms) at 400 hertz (±0.02 percent) to the guidance system. One dc-dc converter supplies regulated ±28 volts dc to the guidance system, while the other dc-dc converter supplies ±28 volts dc to the telemetry system.

Performance

The Agena electrical system voltages and currents were as expected at lift-off, and the system satisfactorily supplied power to all electrical loads throughout the flight.

The battery current load profile was as expected for this mission. The inverter and converter voltages were within specification at lift-off, and remained essentially constant throughout flight. Table VI-II summarizes the electrical system performance.

The inverter frequency is not monitored on Agena vehicles; however, performance of the guidance system indicated the inverter frequency was normal and stable.

TABLE VI-II. - AGENA ELECTRICAL SYSTEM FLIGHT PERFORMANCE SUMMARY, MARINER VENUS 67

Measurement	Units	Tolerance[a]	Measurement number	Flight values at -					
				Lift-off	First ignition	First shutdown	Second ignition	Second shutdown	Spacecraft separation
Pyrotechnic battery voltage	V	22.5 to 29.5	C-141	26.7	26.7	26.7	26.7	26.7	26.7
Main battery voltage	V dc	22.5 to 29.5	C-1	26.2	26.0	26.5	26.5	26.5	26.5
Battery current	A dc	-----------	C-4	15	18	15	18	15	15
Guidance converter, +28.3 V(dc) regulated	V dc	27.7 to 28.9	C-3	28.0	28.0	28.0	28.0	28.0	28.0
Guidance converter, -28.3 V(dc) regulated	V dc	-27.7 to -28.9	C-5	-28.8	-28.8	-28.8	-28.8	-28.8	-28.8
Guidance inverter, phase AB	V ac (rms)	112.7 to 117.3	C-31	113.4	113.4	113.4	113.4	113.4	113.4
Guidance inverter, phase BC	V ac (rms)	112.7 to 117.3	C-32	113.4	113.4	113.4	113.4	113.4	113.4
Telemetry converter 1, +28.3 V(dc) regulated	V dc	27.7 to 28.9	H204	28.0	28.0	28.0	28.0	28.0	28.0

[a]Tolerances listed are for telemetry data. Tolerances listed in electrical system description are equipment specifications.

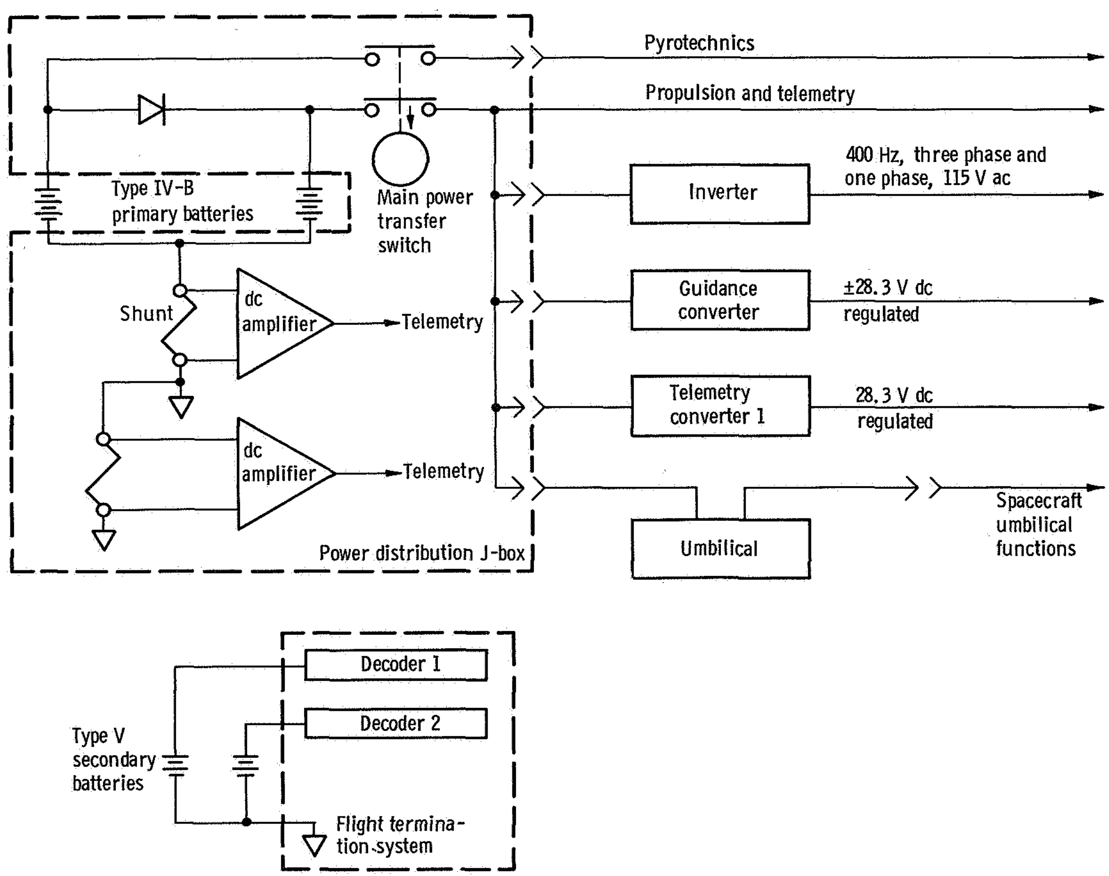


Figure VI-8. - Agena electrical system, Mariner Venus 67.

COMMUNICATION AND CONTROL SYSTEM

by Richard L. Greene

Description

The Agena communication and control system consists of telemetry, tracking, and flight termination subsystems.

The telemetry subsystem is mounted in the forward section. It monitors and transmits the Agena functional and environmental measurements during ascent. The Pulse Amplitude Modulation/Frequency Modulation/Frequency Modulation (PAM/FM/FM) telemetry unit contains a very high frequency (VHF) transmitter, voltage controlled oscillators, a commutator, a switch and calibrate unit, and a dc-dc converter. The transmitter operates on an assigned frequency of 244.3 megahertz at a power output of 10 watts. The telemetry subsystem consists of 10 continuous subcarrier channels and two commutated subcarrier channels.

A total of 61 measurements were telemetered from the Agena vehicle. Appendix B describes the launch vehicle telemetry measurements. Five continuous subcarrier channels were used for accelerometer data; two continuous channels were used for spacecraft V-band tension measurements; one continuous channel was modulated with spacecraft data; one continuous channel monitored the Agena attitude control thrusters; and one continuous channel was time shared by the velocity meter accelerometer and velocity meter counter. The turbine speed signal did not utilize a subcarrier channel but directly modulated the transmitter during engine operation. The remaining 49 measurements were monitored on the two commutated subcarrier channels. These channels are commutated at 5 rps with 60 segments on each channel.

The airborne tracking subsystem includes a C-band beacon transponder, a radio-frequency (RF) switch, and an antenna. The transponder receives coded signals from the tracking radar on a carrier frequency of 5690 megahertz, and transmits coded responses on a carrier frequency of 5765 megahertz at a minimum pulsed power of 200 watts at the input terminals of the antenna. The coded responses are at pulse repetition rates from 0 to 1600 pulses per second. The pulse rate varies inversely with range. The RF switch connects the output of the transponder to either the umbilical for ground checkout or the antenna for flight.

The flight termination subsystem provides a range safety flight termination capability for the Agena, from lift-off through Agena engine first burn. This subsystem consists of two receiver-decoders which are coupled to two antennas by a multicoupler, two batteries, two destruct initiators, and a destruct charge. These units are connected to provide redundant flight termination capability, with the exception of the multicoupler and destruct charge. Flight termination, if necessary, is initiated by the Range Safety Officer through commands from the range safety transmitter. The destruct charge, located near the fuel-oxidizer bulkhead, ruptures both propellant tanks and effects dispersion of the propellants.

The vehicle was configured to permit on-pad conversion to a self-destruct subsystem. This conversion would represent a weight saving of 11.32 kilograms (25 lb) and thereby extend the launch opportunity. However, because the launch occurred during the early portion of the opportunity, the conversion was not required.

Subsystem Performance

Telemetry subsystem. - The telemetry subsystem performance was satisfactory. Stations at Cape Kennedy and Antigua provided telemetry data from lift-off through Agena engine first burn. Ascension provided the coverage for the Agena engine second burn. Telemetry coverage of Agena-spacecraft separation and the Agena posigrade maneuver were provided by the range instrumentation ship Twin Falls. Signal strength data from all stations showed an adequate and continuous signal level from the vehicle telemetry transmitter during these periods. The telemetry data indicated that the performance of the voltage controlled oscillators, switch and calibrate unit, dc-dc converters, and commutator were satisfactory. The telemetry stations that supported the flight and the telemetry data coverage that was provided are shown in appendix C.

Tracking subsystem. - The tracking subsystem performance was satisfactory throughout the flight. The C-band transponder transmitted a continuous response to received interrogations for the required tracking periods. The radar coverage provided by each supporting station is shown in appendix C.

Flight termination subsystem. - The flight termination subsystem maintained the capability to destruct the vehicle throughout the flight. Each receiver signal strength remained well above the airborne receiver threshold from lift-off until the flight termination subsystem was disabled shortly after Agena engine first cutoff.

VII. LAUNCH OPERATIONS

by Frank E. Gue and Alvin C. Hahn

PRELAUNCH ACTIVITIES

The major prelaunch activities at the Eastern Test Range (ETR) for the Mariner Venus 67 launch are shown in table VII-I. All prelaunch tests were completed satisfactorily. The significant schedule delays and problems which occurred during the prelaunch period are as follows:

(1) Prelaunch checkout: The launcher release valve seats were routinely replaced before the Atlas was erected. One of the replaced valve seats failed during checkout and the valve was replaced.

(2) Booster flight acceptance test (B-FACT) 2: Three recycles were required during the Guidance Command Test because of a personnel error at the ground computer.

(3) Joint flight acceptance composite test (J-FACT): The test was postponed for 2 days because spacecraft problems required demating of the encapsulated spacecraft. The test was accomplished without the spacecraft, the spacecraft adapter, and shroud. The initiation signal to separate the Agena drain line umbilicals was not generated because the power source was not turned on. Upon completion of the plus countdown, the power source was turned on and the circuit was verified.

(4) Simulated launch test: The start tank pressurization for the Atlas did not occur as planned at T - 18 seconds because the engine arm switch was in the normal launch position. The count was recycled to T - 7 minutes. Then, with the switch in the proper position, the tanks were successfully pressurized and vented.

COUNTDOWN AND LAUNCH

The Mariner Venus 67 space vehicle was successfully launched from ETR Launch Complex 12 on June 14, 1967. Lift-off, 5.08-centimeter (2-in.) motion, occurred at 0101:00.176 Eastern Standard Time. The countdown proceeded as planned until the scheduled T - 7 minute hold. The hold was extended 14 minutes to change the flight plan from 14F to 14G. Plan 14G provided optimum coverage from the downrange radar and telemetry stations. Launch occurred on the first attempt within 0.2 second of the opening of the window for plan 14G. The performance of the launch vehicle ground equipment during the countdown was satisfactory.

TABLE VII-I. - MAJOR PRELAUNCH ACTIVITIES AT EASTERN TEST RANGE, MARINER VENUS 67

Date	Event
3/17/67	Atlas arrived at Eastern Test Range
3/24/67	Agena arrived at Eastern Test Range
4/14/67	Erection of Atlas
5/02/67	Atlas dual propellant tanking test
5/05/67	Booster flight acceptance test (B-FACT) 1
5/26/67	Booster flight acceptance test (B-FACT) 2
5/29/67	Atlas-Agena mate
5/31/67	Agena-spacecraft mate
6/01/67	Radiofrequency interference (RFI) test
6/02/67	Spacecraft demate
6/07/67	Joint flight acceptance composite test (J-FACT)
6/08/67	Agena-spacecraft mate
6/09/67	Simulated launch test
6/14/67	Launch

VIII. CONCLUDING REMARKS

The Atlas-Agena vehicle met mission requirements as evidenced by the accurate injection of the Mariner onto the Venus transfer ellipse. The Mariner achieved a successful flyby of the planet Venus and obtained valuable scientific data.

A backup method (optical tracking) was employed to acquire the vehicle. Procedure changes were made subsequently to increase the probability of acquisition by the primary "cube acquisition" method.

An Agena engine chamber pressure anomaly occurred that indicate that the interim changes in the propulsion system were inadequate. This emphasizes the necessity for completion of the test program to determine the cause of the anomaly and for the implementation of final recommended solutions.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, January 20, 1969,
491-05-00-02-22.

APPENDIX A

SEQUENCE OF MAJOR FLIGHT EVENTS, MARINER VENUS 67

by Richard L. Greene

Nominal time, sec	Actual time, sec	Event description	Source	Event monitor[a]
0	0	Lift-off (0101:00.176 EST)	-------------	5.08-cm (2-in.) motion switch[b] (M30X)
128.0	128.6	Atlas booster engine cutoff	Atlas guidance	X-axis accelerometer (D7)
297.3	296.7	Atlas sustainer engine cutoff		X-axis accelerometer (D7)
298.7	308.3	Start Agena primary timer		Guidance and control monitor (D14)
317.4	317.9	Atlas vernier engine cutoff		X-axis accelerometer (D7)
		Uncage Agena gyros		Guidance and control monitor (D14)
		Jettison horizon sensor fairings		Guidance and control monitor (D14)
319.5	320.1	Fire nose shroud squibs		Shroud separation (A52)
321.5	322.3	Atlas-Agena separation		Guidance and control monitor (D14)
324.0	324.1	Activate cold gas attitude control	Separation switch	Guidance and control monitor (D14)
347.7	357.4	Initiate -120 deg/min pitch rate	Agena timer	Pitch torque rate (D73)
352.7	362.4	Transfer to -3.2 deg/min pitch		Pitch torque rate (D73)
370.7	380.4	Deactivate pitch and yaw cold gas attitude control		Guidance and control monitor (D14)
		Enable velocity meter and arm engine control		Velocity meter accelerometer (D88)
		Agena engine first-burn ignition squibs		Switch group Z (B13)
371.9	381.6	Agena engine thrust at 90 percent chamber pressure		Chamber pressure (B91)
372.2	381.8	Fire helium pressure squibs (open)		(b)
488.7	498.3	Arm Agena engine shutdown circuit for Agena command destruct		(b)
515.16	525.3	Agena engine first cutoff	Velocity meter	Chamber pressure (B91
515.6	525.3	Activate pitch and yaw cold gas attitude control	Velocity meter	Guidance and control monitor (D14)
523.7	533.4	Close propellant isolation valves	Agena timer	Propellant isolation valve monitor (B130)

[a]All events except those noted are monitored on Agena telemetry. The designation in parentheses is the monitor measurement designation. See appendix B for measurement range and channel assignment.
[b]No direct measurement.

Nominal time, sec	Actual time, sec	Event description	Source	Event monitor[a]
539.7	549.4	Transfer to -420 deg/min pitch rate	Agena timer	Pitch torque rate (D73)
		Disarm Agena command destruct		(b)
551.7	561.3	Fire 0° horizon sensor position squibs		Pitch horizon sensor (D41)
653.5	663.1	Fire helium valve squib 2 (close)		(b)
1312.1	1318.8	Transfer to second-burn velocity increment		Velocity meter counter (D83)
1316.1	1317.8	Enable velocity meter		Velocity meter accelerometer (D88)
		Open propellant isolation valves		Propellant isolation valve monitor (B130)
1318.1	1319.8	Deactivate pitch and yaw cold gas attitude control		Guidance and control monitor (D14)
		Fire Agena second-burn squibs		Switch group Z (D13)
1319.3	1320.9	Agena engine thrust at 90 percent chamber pressure		Chamber pressure (B91)
1409.1	1410.7	Arm engine shutdown circuit		(b)
1414.9	1415.3	Agena engine second cutoff	Velocity meter	Chamber pressure (B91)
		Pitch and yaw cold gas attitude control on	Velocity meter	Guidance and control monitor (D14)
1425.1	1426.7	Disable velocity meter and switch telemetry to velocity meter counter output	Agena timer	Velocity meter counter (D83)
1575.1	1576.7	Fire payload ejection squibs		Spacecraft adapter V-band (A105)
1578.1	1579.7	Initiate +180 deg/min yaw rate		Yaw torque rate (D51)
1587.1	1588.7	Remove yaw program		Yaw torque rate (D51)
		Transfer to +5.0 deg/min pitch rate		Pitch torque rate (D73)
1875.1	1876.8	Initiate posigrade maneuver		Velocity meter accelerometer (D88)

[a]All events except those noted are monitored on Agena telemetry. The designation in parentheses is the monitor measurement designation. See appendix B for measurement range and channel assignment.
[b]No direct measurement.

APPENDIX B

LAUNCH VEHICLE INSTRUMENTATION SUMMARY, MARINER VENUS 67

by Edwin S. Jeris and Richard L. Greene

TABLE B-I. - ATLAS TELEMETRY

Measurement number	Description	Channel assignment[a]	Measurement range, low/high: SI Units	Measurement range, low/high: U.S. Customary Units
A743T	Ambient temperature at sustainer instrumentation panel	11-41	227.5 to 561 K	-50° to 550° F
A745T	Ambient temperature at sustainer fuel pump	11-45	227.5 to 561 K	-50° to 550° F
D1V	Range safety command cutoff output	5-S	(b)	
D1V	Range safety command cutoff output	15-1	0 to 5 V dc	
D7V	Number 1 range safety command radiofrequency input automatic gain control	15-3	0 to 10 000 μV	
D3X	Range safety command destruct output	16-S	0 to 6 V dc	
E28V	Main dc voltage	18-1/31	20 to 35 V dc	
E51V	400-Hz ac phase A	18-11	105 to 125 V dc	
E52V	400-Hz ac phase B	18-29	105 to 125 V ac	
E53V	400-Hz ac phase C	18-41	105 to 125 V ac	
E95V	28-V dc guidance power input	13-15	20 to 35 V dc	
E96V	115-V ac 400-Hz phase A to guidance	13-37	105 to 125 V ac	
E151V	400-Hz phase A waveform	10	0 to 150 V ac	
F1P	Liquid oxygen tank helium pressure, absolute	15-9	0 to 34.5 N/cm^2	0 to 50 psi
F3P	Fuel tank helium pressure, absolute	15-11	0 to 68.9 N/cm^2	0 to 100 psi
F116P	Differential pressure across bulkhead	18-13/43	0 to 17.2 N/cm^2	0 to 25 psi
F125P	Booster control pneumatic regulator output pressure, absolute	13-21	0 to 689 N/cm^2	0 to 1000 psi
F246P	Booster tank helium bottle pressure, absolute	13-55	0 to 2413 N/cm^2	0 to 3500 psi
F288P	Start pneumatic regulator output	13-1	0 to 551.5 N/cm^2	0 to 800 psi
F291P	Sustainer control helium bottle	13-3	0 to 2413 N/cm^2	0 to 3500 psi
F247T	Booster tank helium bottle temperature	11-31	33.5 to 116.5 K	-400° to 250° F
G4C	Pulse beacon magnetron average current	15-15	0 to 5 V dc	
G82E	Rate beacon radiofrequency output	15-17	0 to 5 V dc	
G3V	Pulse beacon automatic gain control	15-19	0 to 5 V dc	
G279V	Rate beacon automatic gain control 1	15-21	0 to 5 V dc	
G280V	Rate beacon automatic gain control 2	15-13	0 to 5 V dc	
G282V	Rate beacon phase detector 1	15-45	0 to 5 V dc	

[a]First number indicates Interrange Instrumentation Group (IRIG) subcarrier channel used. Second number(s) indicates commutated position(s) for measurement. If no second number is indicated, the channel is used continuously for the designated transducer.

[b]Items are determined from a step change in voltage.

TABLE B-I. - Continued. ATLAS TELEMETRY

Measurement number	Description	Channel assignment[a]	Measurement range, low/high	
			SI Units	U.S. Customary Units
G287V	Decoder pitch output	15-47	0 to 5 V dc	
G288V	Decoder yaw output	15-49	0 to 5 V dc	
G296V	Pulse beacon 15-V dc power supply	13-9	0 to 5 V dc	
G293V	Decoder 10-V dc power supply	13-13	0 to 5 V dc	
G354V	Rate beacon 25- to 30-V dc power supply	13-11	0 to 5 V dc	
G590V	Discrete binary 1	16-33	0 to 5 V dc	
G591V	Discrete binary 2	16-35	0 to 5 V dc	
G592V	Discrete binary 4	16-37	0 to 5 V dc	
G593V	Discrete binary 8	16-39	0 to 5 V dc	
G363X	Jettison shroud	6-S	(b)	
G364X	Start Agena restart timer	2	(b)	
H3P	Booster hydraulic pump discharge pressure, absolute	13-41	0 to 2413 N/cm^2	0 to 3500 psi
H33P	B1 hydraulic accumulator pressure, absolute	15-31	0 to 2413 N/cm^2	0 to 3500 psi
H130P	Sustainer hydraulic pump discharge pressure, absolute	15-33	0 to 2413 N/cm^2	0 to 3500 psi
H140P	Sustainer-vernier hydraulic pressure, absolute	15-35	0 to 2413 N/cm^2	0 to 3500 psi
H224P	Booster hydraulic system low pressure, absolute	15-7	0 to 413.6 N/cm^2	0 to 600 psi
H601P	Sustainer hydraulic return line	18-7/37	0 to 413.6 N/cm^2	0 to 600 psi
M79A	Vehicle axial accelerometer fine	7	-0.5 to 0.5 g's	
M30X	Vehicle 2-inch (5.08-cm) motion	7-S	(b)	
M32X	Conax valve command	5-S	(b)	
P83B	Booster 2 pump speed	15-41	4000 to 7000 rpm	
P84B	Booster 1 pump speed	4	6000 to 6950 rpm	
P349B	Sustainer pump speed	3	9900 to 11 200 rpm	
P529D	Sustainer main liquid oxygen valve	13-43	0° to 90°	
P330D	Sustainer fuel valve position	13-35	23° to 54°	
P1P	Booster 1 liquid oxygen pump inlet pressure, absolute	18-9	0 to 103.4 N/cm^2	0 to 150 psi
P2P	Booster 1 fuel pump inlet pressure, absolute	13-31	0 to 68.9 N/cm^2	0 to 100 psi
P6P	Sustainer thrust chamber pressure, absolute	18-3/33	0 to 689 N/cm^2	0 to 1000 psi
P26P	Booster liquid oxygen regulator reference pressure, absolute	13-17	344.7 to 689 N/cm^2	500 to 1000 psi
P27P	Vernier fuel tank pressure, absolute	13-39	0 to 689 N/cm^2	0 to 1000 psi
P28P	Vernier 1 thrust chamber pressure, absolute	18-15	0 to 275.8 N/cm^2	0 to 400 psi

[a]First number indicates Interrange Instrumentation Group (IRIG) subcarrier channel used. Second number (s) indicates commutated position(s) for measurement. If no second number is indicated, the channel is used continuously for the designated transducer.

[b]Items are determined from a step change in voltage.

TABLE B-I. - Continued. ATLAS TELEMETRY

Measure-ment number	Description	Channel assign-ment[a]	Measurement range, low/high	
			SI Units	U.S. Customary Units
P29P	Vernier 2 thrust chamber pressure, absolute	18-17	0 to 275.8 N/cm^2	0 to 400 psi
P30P	Vernier liquid oxygen tank pressure, absolute	13-53	0 to 689 N/cm^2	0 to 1000 psi
P47P	Vernier 1 liquid oxygen inlet pressure, absolute	13-45	0 to 413.7 N/cm^2	0 to 600 psi
P49P	Vernier 1 fuel inlet pressure, absolute	13-49	0 to 413.7 N/cm^2	0 to 600 psi
P55P	Sustainer fuel pump inlet pressure, absolute	13-5	0 to 68.9 N/cm^2	0 to 100 psi
P56P	Sustainer liquid oxygen pump inlet pressure, absolute	18-5	0 to 103.4 N/cm^2	0 to 150 psi
P59P	Booster 2 thrust chamber pressure, absolute	18-19	0 to 511.6 N/cm^2	0 to 800 psi
P60P	Booster 1 thrust chamber pressure, absolute	18-21	0 to 551.6 N/cm^2	0 to 800 psi
P100P	Gas generator combustion chamber pressure, absolute	15-51	0 to 413.7 N/cm^2	0 to 600 psi
P330P	Sustainer fuel pump discharge pressure, absolute	15-55	0 to 1034.2 N/cm^2	0 to 1500 psi
P339P	Sustainer gas generator discharge pressure, absolute	18-55	0 to 551.6 N/cm^2	0 to 800 psi
P334P	Sustainer liquid oxygen regulator reference pressure, absolute	13-19	344.7 to 689 N/cm^2	500 to 1000 psi
P15T	Engine compartment air temperature	11-35	227.5 to 561 K	-50° to 550° F
P16T	Engine compartment component temperature	11-55	255.5 to 477.5 K	0° to 400° F
P117T	Booster 2 fuel pump inlet	11-53	255.5 to 311 K	0° to 100° F
P530T	Sustainer liquid oxygen pump inlet temperature	11-1	89 to 105.5 K	-300° to -270° F
P671T	Thrust section ambient temperature quadrant 4	11-15	227.5 to 561 K	-50° to 550° F
P77X	Vernier cutoff relay	8-S	(b)	
P347X	System cutoff relay	8-S	(b)	
P616X	Booster flight lock-in relay	16-19	(b)	
S61D	Roll displacement gyro signal	15-29	-3° to 3°	
S62D	Pitch displacement gyro signal	15-37	-3° to 3°	
S63D	Yaw displacement gyro signal	15-39	-3° to 3°	
S252D	Booster 1 yaw roll	16-15	-6° to 6°	
S253D	Booster 2 yaw roll	16-55	-6° to 6°	
S254D	Booster 1 pitch	7	-6° to 6°	

[a]First number indicates Interrange Instrumentation Group (IRIG) subcarrier channel used. Second number (s) indicates commutated position(s) for measurement. If no second number is indicated, the channel is used continuously for the designated transducer.

[b]Items are determined from a step change in voltage.

TABLE B-I. - Concluded. ATLAS TELEMETRY

Measurement number	Description	Channel assignment[a]	Measurement range, low/high	
			SI Units	U.S. Customary Units
S255D	Booster 2 pitch	16-1	-6° to 6°	
S256D	Sustainer yaw	16-41	-4° to 4°	
S257D	Sustainer pitch	16-45	-4° to 4°	
S258D	Vernier 1 pitch roll	16-3	-70° to 70°	
S259D	Vernier 2 pitch roll	16-5	-70° to 70°	
S260D	Vernier 1 yaw	16-7	-55° to 5°	
S261D	Vernier 2 yaw	16-9	-5° to 55°	
S52R	Roll rate gyro signal	9	-8 to 8 deg/sec	
S53R	Pitch rate gyro signal	8	-6 to 6 deg/sec	
S54R	Yaw rate gyro signal	5	-6 to 6 deg/sec	
S190V	Pitch gyro torque amplifier	15-43	-1 to 1 V ac	
S209V	Programmer 28-V dc test	6	20 to 35 V dc	
S236X	Booster cutoff discrete	9-S	(b)	
S241X	Sustainer cutoff discrete	9-S	(b)	
S245X	Vernier cutoff discrete	9-S	(b)	
S248X	Release payload discrete	9-S	(b)	
S290X	Programmer output Spare Booster jettison Enable discretes	16-29	0 to 28 V dc	
S291X	Programmer output Booster engine cutoff Sustainer engine cutoff Vernier engine cutoff	16-31	0 to 28 V dc	
S359X	Booster staging backup	5-S	(b)	
S384X	Spin motor test output	15-5	0 to 5 V dc	
U101A	Axial acceleration	12	0 to 10 g's	
U80P	Liquid oxygen tank head pressure, differential	16-11	0 to 3.4 N/cm^2	0 to 5 psi
U81P	Fuel tank head pressure, differential	16-13	0 to 1.7 N/cm^2	0 to 2.5 psi
U112V	Acoustica counter output	15-23/53	0 to 5 V dc	
U113V	Acoustica valve position feedback	13-33	0 to 5 V dc	
U132V	Acoustica station counter output	13-7	0 to 5 V dc	
U134V	Acoustica time shared oscillator output	18-23/53	0 to 5 V dc	
U135V	Acoustica sensor signal	18-39	0 to 5 V dc	
U605V	Acoustica time shared integrator switch	18-35	0 to 5 V dc	
Y44P	Interstage adapter pressure, absolute	13-23	0 to 10.3 N/cm^2	0 to 15 psi
Y45T	Interstage adapter temperature	11-5	144 to 366.5 K	-200° to 200° F
Y41X	Start Agena primary timer	5-S	(b)	

[a]First number indicates Interrange Instrumentation Group (IRIG) subcarrier channel used. Second number(s) indicates commutated position(s) for measurement. If no second number is indicated, the channel is used continuously for the designated transducer.

[b]Items are determined from a step change in voltage.

TABLE B-II. - AGENA TELEMETRY

Measurement number	Description	Channel assignment[a]	Measurement range, low/high	
			SI Units	U.S. Customary Units
A52	Shroud separation	15-44	(b)	
A105	Spacecraft V-band tension 1	12	0 to 13.34×10^3 N	0 to 3000 lb
A106	Spacecraft V-band tension 2	13	0 to 13.34×10^3 N	0 to 3000 lb
AT106	Spacecraft Agena diaphragm temperature	16-18	255 to 345 K	0° to 160° F
B1	Fuel pump inlet pressure, gage	15-15	0 to 69 N/cm^2	0 to 100 psi
B2	Oxidizer pump inlet pressure, gage	15-17	0 to 1034 N/cm^2	0 to 100 psi
B11	Oxidizer venturi inlet pressure, absolute	15-19/49	0 to 1034 N/cm^2	0 to 1500 psi
B12	Fuel venturi inlet pressure, absolute	15-23/53	0 to 1034 N/cm^2	0 to 1500 psi
B13	Switch group Z	15-7/22/37/53	255 to 311 K	0° to 100° F
B31	Fuel pump inlet temperature	15-6	255 to 311 K	0° to 100° F
B32	Oxidizer pump inlet temperature	15-8	255 to 311 K	0° to 100° F
B35	Turbine speed		(c)	
B91	Combustion chamber pressure, gage	15-4/34	328 to 379 N/cm^2	475 to 550 psi
B130	Propellant isolation valve monitor	15-11/13/23/27/31/33/41/48/51/53/56	(b)	
C1	28-V dc unregulated supply	16-40	22 to 30 V dc	
C3	28-V dc regulator (guidance and control) 1	15-12	22 to 30 V dc	
C4	28-V dc unregulated current	16-13/44	0 to 100 A	
C5	-28-V dc regulator (guidance and control)	15-30	-30 to -22 V dc	
C21	400-Hz, three-phase; inverter temperature	15-14	255 to 367 K	0° to 200° F
C31	400-Hz, three-phase; bus phase AB	15-18	90 to 130 V ac	
C32	400-Hz, three-phase; bus phase BC	15-20	90 to 130 V ac	

[a]First number indicates Interrange Instrumentation Group (IRIG) subcarrier channel used. Second number(s) indicates commutated position(s) for measurement. If no second number is indicated, the channel is used continuously for the designated transducer.

[b]Items are determined from a step change in voltage.

[c]Turbine speed signal does not utilize a subcarrier channel but directly modulates transmitter during engine operation.

TABLE B-II. - Continued. AGENA TELEMETRY

Measurement number	Description	Channel assignment[a]	Measurement range, low/high	
			SI Units	U.S. Customary Units
C38	Structure current monitor	15-10/25/40/55	0 to 50 A	
C141	Pyrotechnic bus voltage	15-5/35	22 to 30 V dc	
D7	X-axis accelerometer	11	-4 to 12 g's	
D8	Y-axis accelerometer	8	-1.5 to 1.5 g's	
D9	Z-axis accelerometer	9	-1.5 to 1.5 g's	
D14	Guidance and control monitor	16-27	22 to 30 V dc	
D41	Horizon sensor pitch	16-45	-5° to 5°	
D42	Horizon sensor roll	16-46	-5° to 5°	
D46	Gas valve temperature cluster 1	15-39	228 to 339 K	-50° to 150° F
D47	Gas valve temperature cluster 2	15-36	228 to 339 K	-50° to 150° F
D51	Yaw torque rate (ascent mode)	16-38	-200 to 200 deg/min	
D51	Yaw torque rate (orbital mode)	16-38	-10 to 10 deg/min	
D54	Horizon sensor head temperature (right head)	15-47	228 to 367 K	-50° to 200° F
D55	Horizon sensor head temperature (left head)	15-46	228 to 367 K	-50° to 200° F
D59	Control gas supply pressure (high), absolute	16-47	0 to 2758 N/cm^2	0 to 4000 psi
D60	Hydraulic oil pressure, absolute	15-21	0 to 2758 N/cm^2	0 to 4000 psi
D66	Roll torque rate (ascent mode)	16-41	-50 to 50 deg/min	
D66	Roll torque rate (orbital mode)	16-41	-4 to 4 deg/min	
D68	Pitch actuator position	15-3	-2.5 to 2.5 deg	
D69	Yaw actuator position	15-24	-2.5 to 2.5 deg	
D70	Control gas supply temperature	15-42	228 to 367 K	-50° to 200° F
D72	Pitch gyro output (ascent mode)	16-36	-10° to 10°	-50° to 200° F
D72	Pitch gyro output (orbital mode)	16-36	-5° to 5°	
D73	Pitch torque rate (ascent mode)	15-35	-200 to 200 deg/min	
D73	Pitch torque rate (orbital mode)	16-35	-10 to 10 deg/min	

[a]First number indicates Interrange Instrumentation Group (IRIG) subcarrier channel used. Second number(s) indicates commutated position(s) for measurement. If no second number is indicated, the channel is used continuously for the designated transducer.

TABLE B-II. - Concluded. AGENA TELEMETRY

Measurement number	Description	Channel assignment[a]	Measurement range, low/high	
			SI Units	U.S. Customary Units
D74	Yaw gyro output (ascent mode)	16-39	-10° to 10°	
D74	Yaw gyro output (orbital mode)	16-39	-5° to 5°	
D75	Roll gyro output (ascent mode)	16-42	-10° to 10°	
D75	Roll gyro output (orbital mode)	16-42	-5° to 5°	
D83	Velocity meter acceleration	14	0 to 2000 pulses/sec	
D86	Velocity meter cutoff switch	16-28	(b)	
D88	Velocity meter counter	14	Binary code (50 pulses/sec)	
D129	Inertial reference package internal case temperature	15-54	255.5 to 341.5 K	0° to 155° F
D149	Gas valves 1 to 6, current	7	(d)	
H47	Beacon receiver pulse repetition rate	15-27	0 to 1600 pulses/sec	
H48	Beacon transmitter pulse repetition rate	15-28	0 to 1600 pulses/sec	
H101	Safe-arm-fire destruct 1	16-2	(b)	
H103	Safe-arm-fire destruct 2	16-4	(b)	
H204	dc-dc Converter 2	16-50	22 to 30 V dc	
H218	Telemetry transmitter temperature	16-49	283 to 350 K	50° to 170° F
H354	Destruct receiver 1 signal level	16-6	0 to 40	
H364	Destruct receiver 2 signal level	16-8	0 to 40	
PL3	Shroud pressure internal, absolute	16-16	0 to 10.32 N/cm^2	0 to 15 psi
PL6	Axial vibration (spacecraft adapter)	18	-30 to 30 g's	
PL25	Axial vibration (spacecraft)	17	-20 to 20 g's	
	JPL base band (spacecraft data)	F	50 bits/sec	

[a]First number indicates Interrange Instrumentation Group (IRIG) subcarrier channel used. Second number(s) indicates commutated position(s) for measurement. If no second number is indicated, the channel is used continuously for the designated transducer.

[b]Items are determined from a step change in voltage.

[d]A unique voltage level is associated with any one or combination of gas valve (jet) activity.

APPENDIX C

TRACKING AND DATA ACQUISITION, MARINER VENUS 67

by Richard L. Greene

The launch vehicle trajectory projected on a world map is shown in figure C-1. A total of 13 tracking and telemetry stations provided radar data (vehicle position, velocity, and acceleration) and telemetry data (vehicle subsystem performance) during the near earth portion of the mission. Coverage was provided by the Eastern Test Range (ETR) uprange stations at Cape Kennedy, Grand Bahama Island, Grand Turk Island, and Antigua and the ETR downrange stations at Ascension, Pretoria (South Africa), and two range instrumentation ships (RIS). Supplementing the ETR downrange station coverage were the NASA Manned Space Flight Network (MSFN) stations at Bermuda, Tananarive (Malagasy Republic), and Carnarvon (Australia).

Telemetry Data

Telemetry data from the Atlas-Agena launch vehicle were recorded on magnetic tape by telemetry stations during all Atlas and Agena engine operations, Agena/spacecraft separation, and postseparation Agena posigrade maneuver. The data were used for postflight analyses of launch vehicle performance. Real-time monitoring of specific Atlas and Agena parameters was provided for verification of occurrence of significant flight events. The submarine cable between the ETR uprange stations permitted real-time monitoring of vehicle telemetered signals through Agena engine first cutoff. The subsequent flight events were monitored by the ETR downrange stations and by the MSFN stations. The occurrence of these events was reported back to Cape Kennedy in "near" real time by ETR single sideband radio links and the NASA Communication Network (NASCOM) voice and data circuits. Figure C-2 shows the specific telemetry coverage provided by each telemetry station.

Radar Data

C-band radar data (time, elevation, azimuth, and range) were provided for real-time operations and postflight analyses. Real-time radar data were provided for monitoring the launch vehicle flight performance for range safety purposes and for assisting the downrange stations in acquiring track of the vehicle. These data were also used for computation of parking orbit elements, injection conditions at the Agena engine first cutoff, injection conditions at Agena engine second cutoff, and Agena final orbit elements. The specific radar coverage provided by each tracking station is presented in figure C-3.

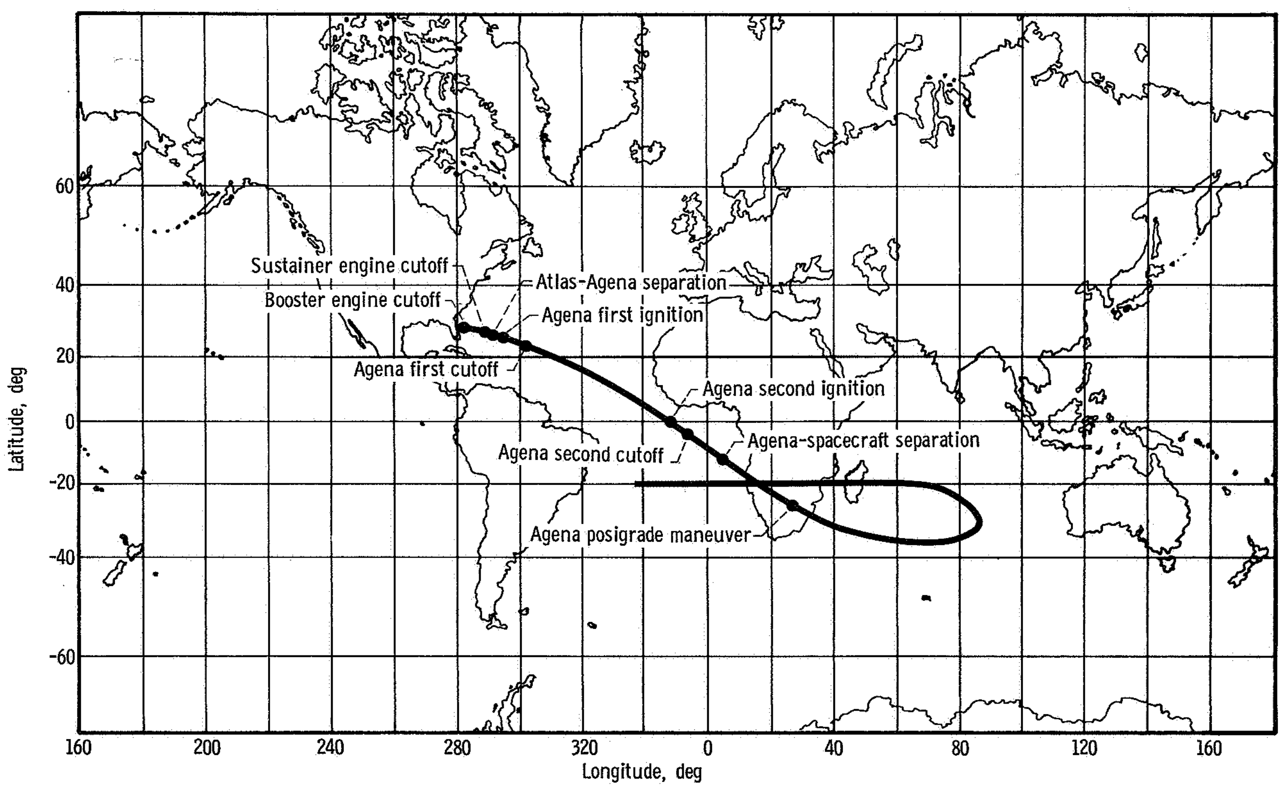


Figure C-1. - Ground trace for Mariner Venus 67.

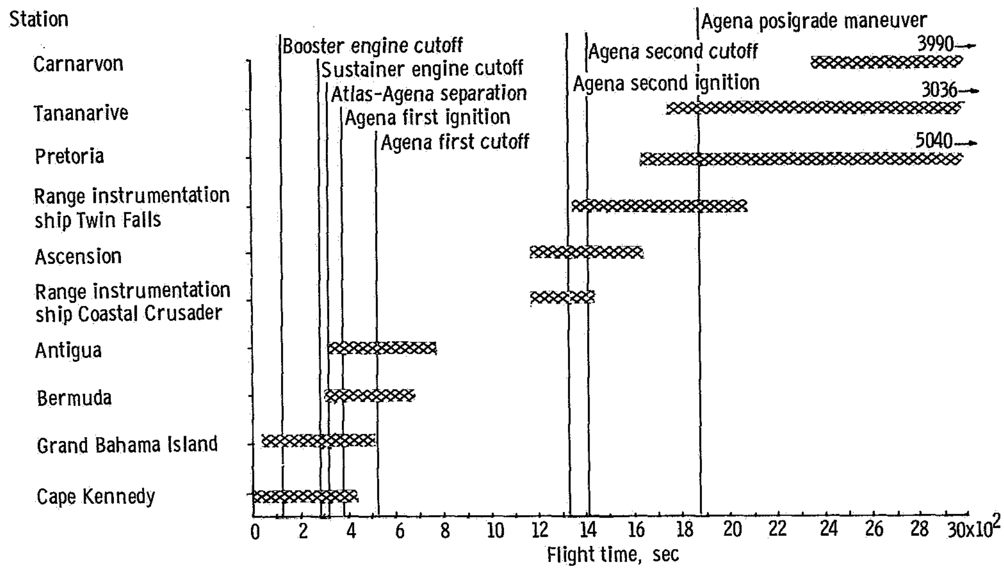


Figure C-2. - Launch vehicle telemetry coverage, Mariner Venus 67.

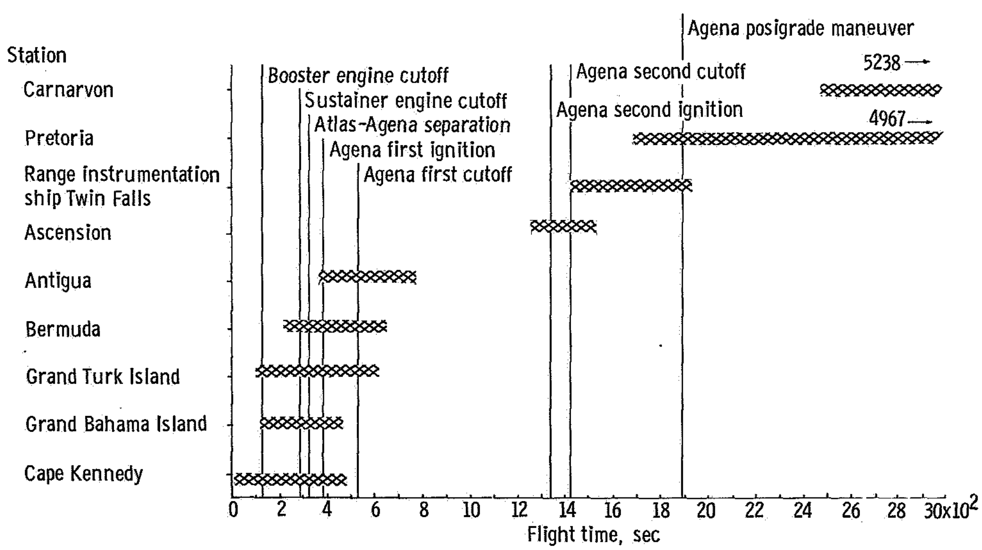


Figure C-3. - Launch vehicle radar coverage, Mariner Venus 67.

APPENDIX D

VEHICLE FLIGHT DYNAMICS, MARINER VENUS 67

by Robert W. York

Flight dynamic data were obtained from three accelerometers installed in the Agena forward section, from one vibration transducer on the spacecraft adapter, and from one vibration transducer mounted on the spacecraft. A summary of dynamic instrumentation locations and characteristics is presented in figure D-1.

The following table presents the actual flight times during which significant dynamic disturbances were recorded.

Event causing disturbance	Time of dynamic disturbance, seconds after lift-off
Lift-off	0
Transonic region	40 to 60
Booster engine cutoff (BECO)	128.80
Sustainer engine cutoff (SECO)	296.84
Horizon sensor fairing jettison	317.90
Shroud separation	319.97
Atlas-Agena separation	322.08
Agena engine first ignition	381.61
Agena engine first cutoff	525.33
Agena engine second ignition	1320.86
Agena engine second cutoff	1415.37

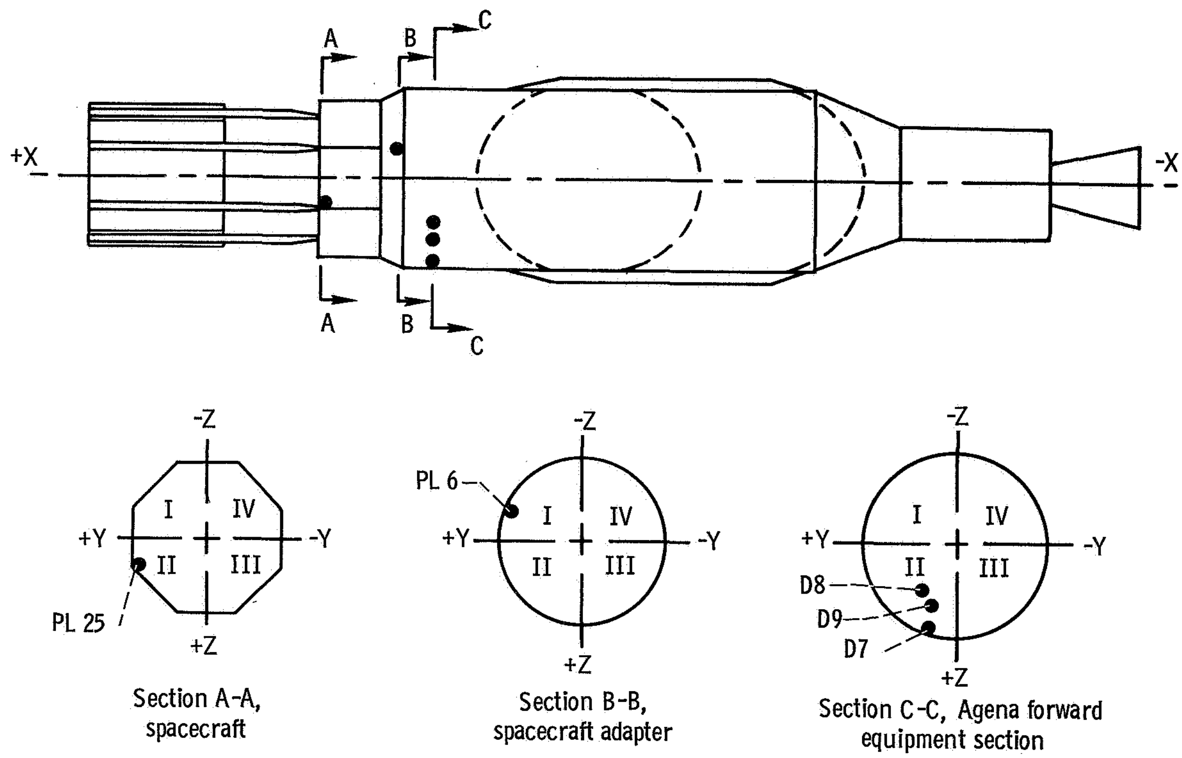
Table D-I summarizes the maximum acceleration levels and corresponding frequencies experienced during significant flight events. All acceleration levels are shown in g's zero-to-peak.

Environmental data recorded for the dynamic disturbances associated with flight events are presented in figures D-2 to D-12.

TABLE D-I. - SUMMARY OF DYNAMIC ENVIRONMENT, MARINER VENUS 67

Event causing disturbance	Time of dynamic disturbance, seconds after lift-off	Accelerometers						Vibrometers			
		Channel 8		Channel 9		Channel 11		Channel 17		Channel 18	
		Measurement D-8 Y axis		Measurement D-9 Z axis		Measurement D-7 X axis		Measurement PL25 Axial		Measurement PL6 Axial	
		Frequency, Hz	g's, zero to peak	Frequency, Hz	g's, zero to peak	Frequency, Hz	g's, zero to peak	Frequency, Hz	g's zero to peak	Frequency, Hz	g's zero to peak
Lift-off	0	45	0.8	100	0.3	5	0.1	5	0.5	5	0.5
Transonic region	40 to 60	45	1.0	96	.2	350	.5	(a)	2.0	(a)	5.0
Booster engine cutoff (BECO)	128.80	45	1.2	4	.7	70	.4	90	.5	100	.5
Sustainer engine cutoff (SECO)	296.84	48	.4	94	.2	94	.4	94	.5	94	.5
Horizon sensor fairing jettison	317.90	52	.3	92	.8	92	4.3	88	.5	110	.5
Shroud separation	319.97	46	.1	92	.1	340	1.1	(a)	18	(a)	Over 30
Atlas-Agena separation	322.08	70	.2	94	.1	340	.4	100	.5	(a)	Over 30
Agena engine first ignition	381.61	48	.7	48	.3	70	.2	80	.5	80	.5
Agena engine first cutoff	525.33	40	.9	90	.3	74	1.3	80	2.5	90	1.5
Agena engine second ignition	1320.86	52	1.1	82	.4	74	1.0	74	1.0	86	.1
Agena engine second cutoff	1415.37	48	1.8	76	.6	76	1.5	76	3.0	72	2.0

[a]Frequency greater than 800 Hz.



Measurement number	Channel number	Measurement location and instrument	Vehicle station	Frequency response, Hz	Range of g's
D8	8	Y-axis accelerometer	257	0 to 35	±1.5
D9	9	Z-axis accelerometer	257	0 to 110	±1.5
D7	11	X-axis accelerometer	257	0 to 160	-4 to 12
PL25	17	Spacecraft axial vibrometer	219.5	20 to 1500	±20
PL6	18	Spacecraft adapter axial vibrometer	245	20 to 2000	±30

Figure D-1. - Dynamic flight instrumentation for Mariner Venus 67.

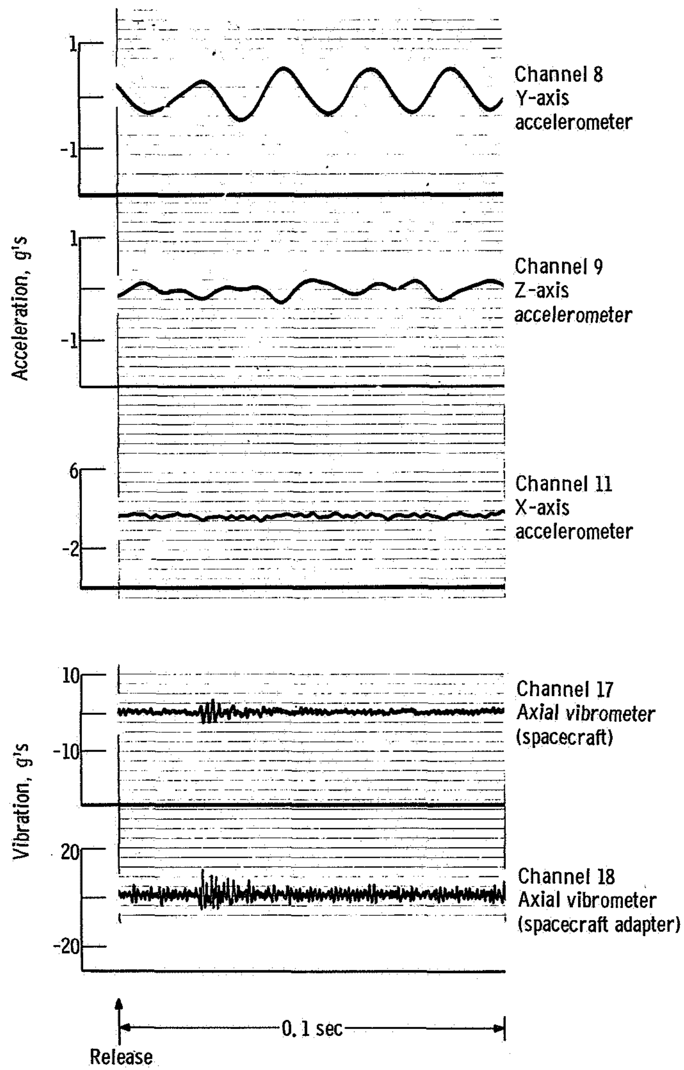


Figure D-2. - Dynamic data at lift-off, Mariner Venus 67.

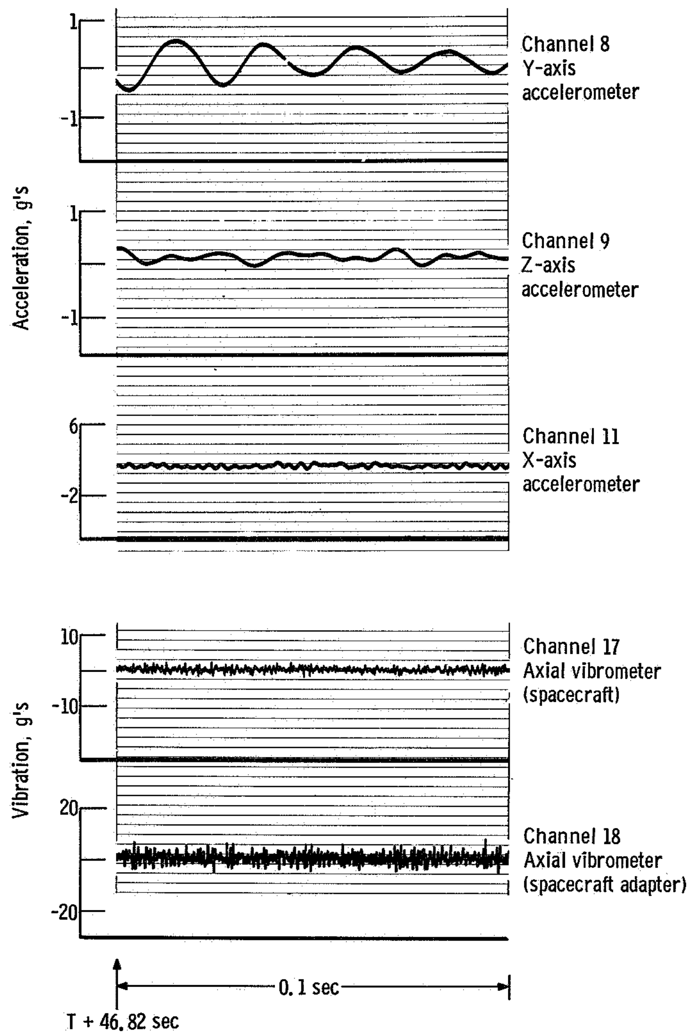


Figure D-3. - Dynamic data during transonic period, Mariner Venus 67.

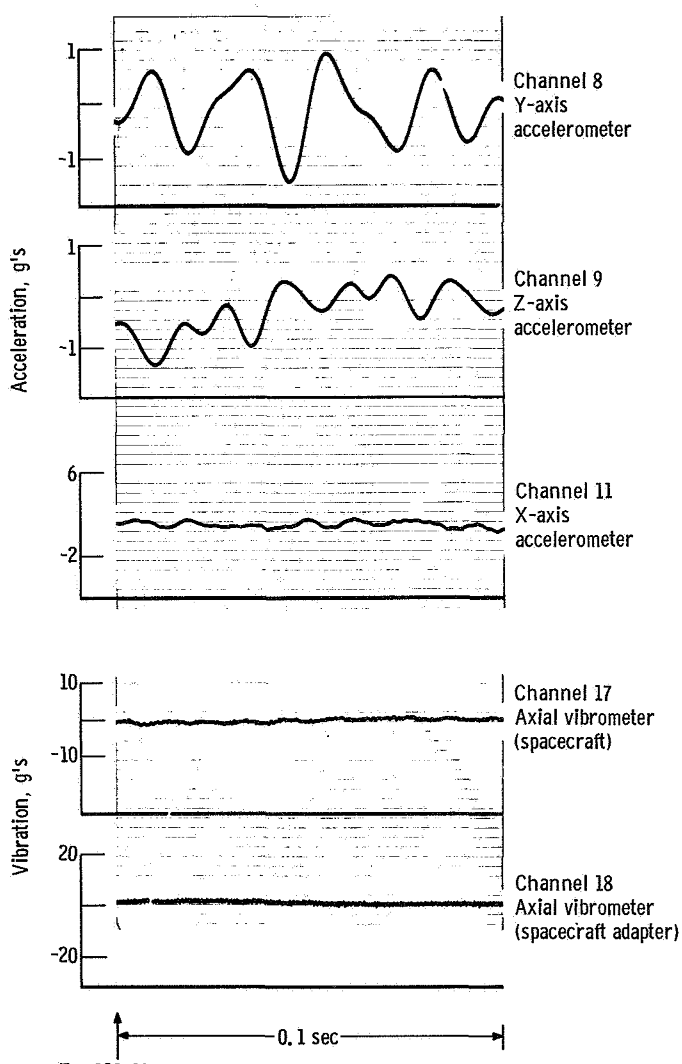


Figure D-4. - Dynamic data near time of booster engine cutoff, Mariner Venus 67.

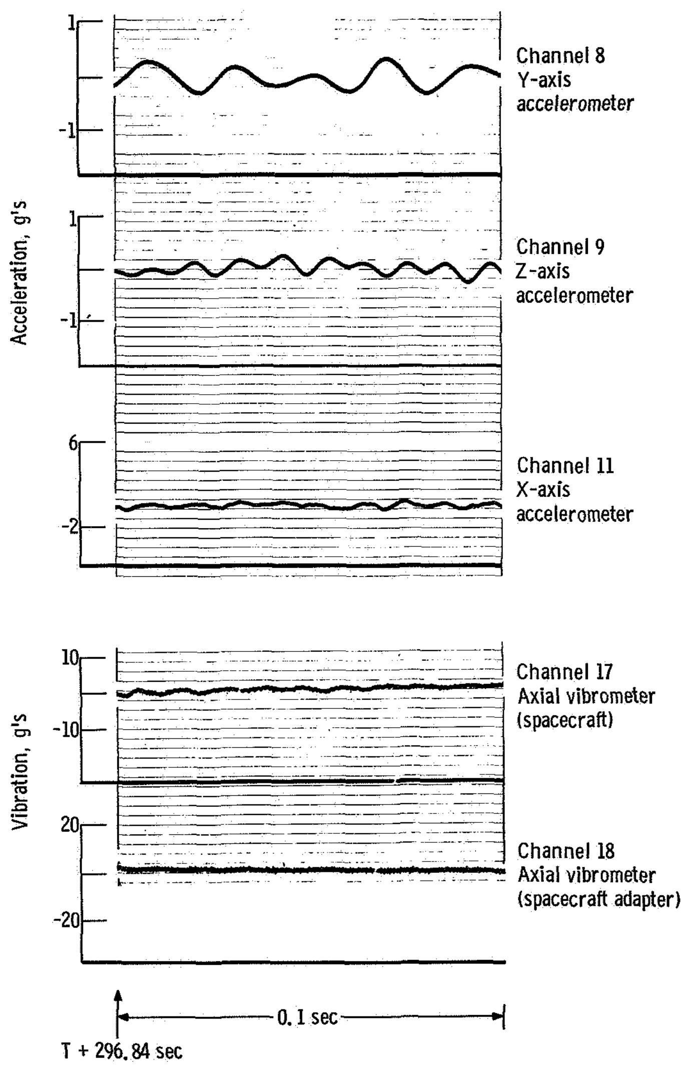


Figure D-5. - Dynamic data near time of sustainer engine cutoff, Mariner Venus 67.

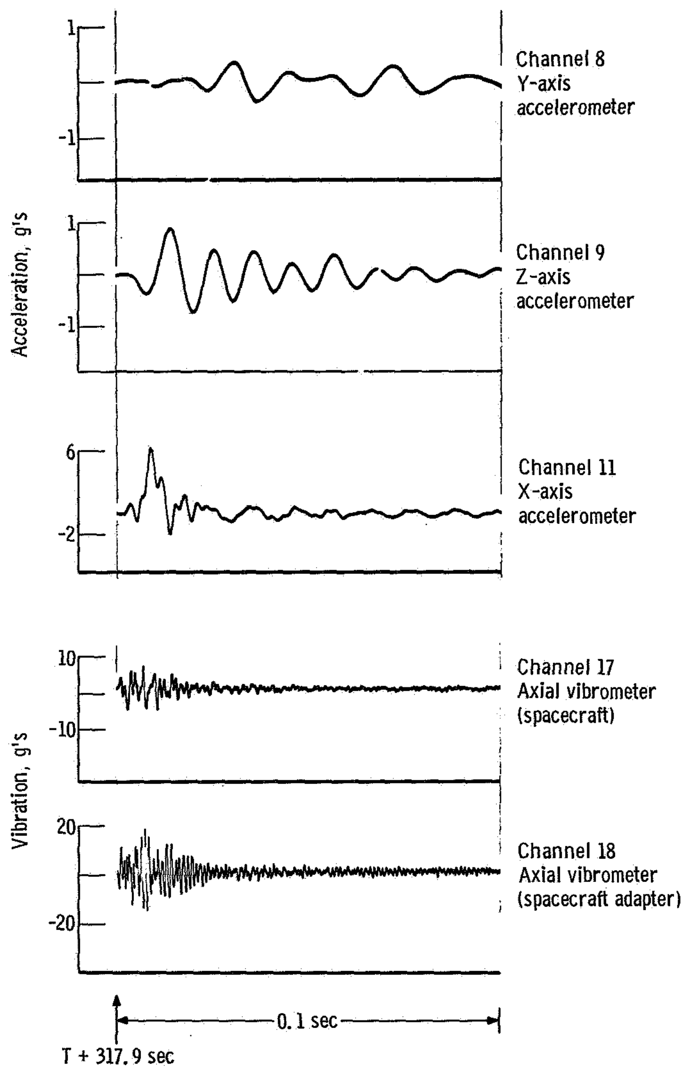


Figure D-6. - Dynamic data at time of horizon sensor fairing jettison, Mariner Venus 67.

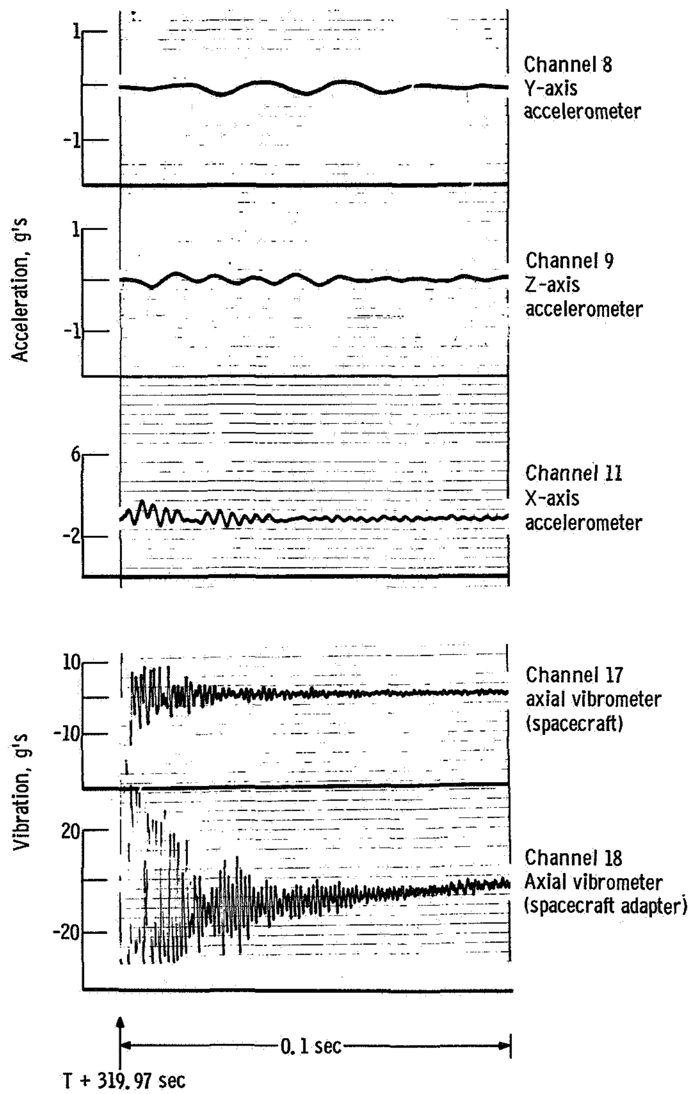


Figure D-7. - Dynamic data near time of shroud separation, Mariner Venus 67.

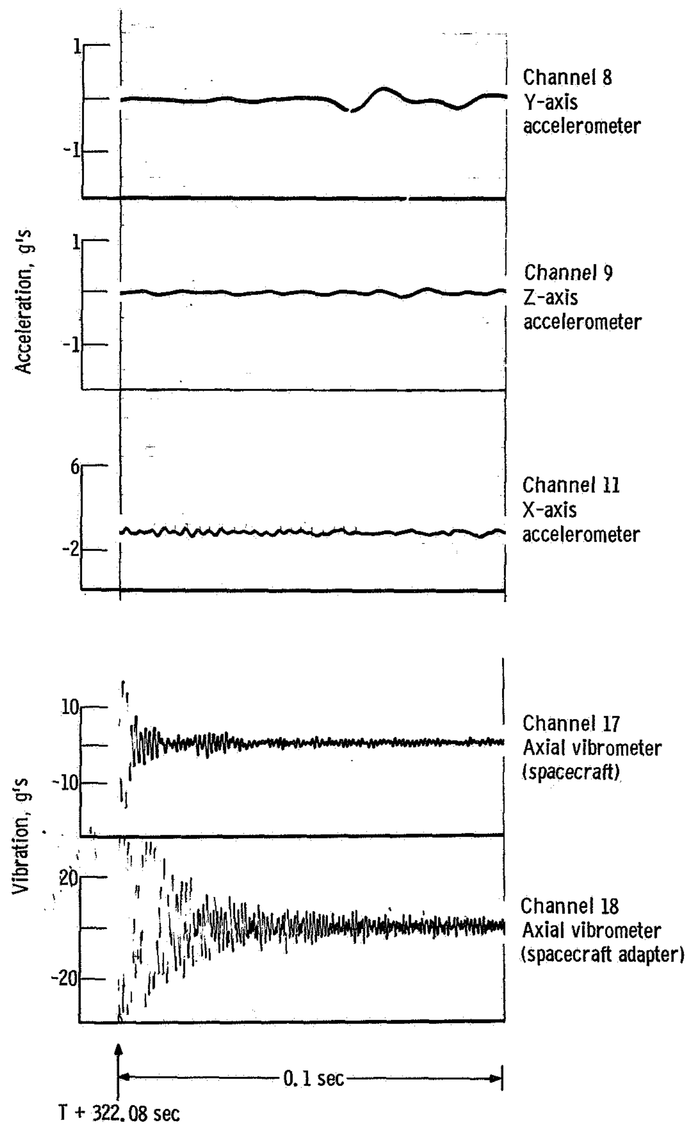


Figure D-8. - Dynamic data near time of Atlas-Agena separation, Mariner Venus 67.

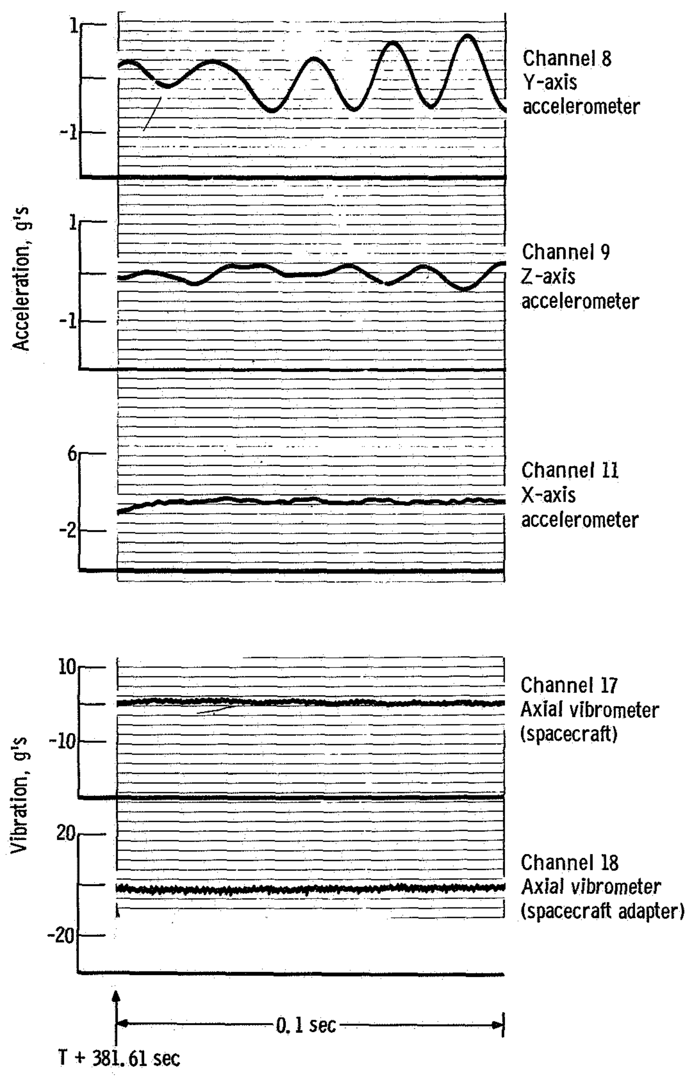


Figure D-9. - Dynamic data near time Agena engine first ignition, Mariner Venus 67.

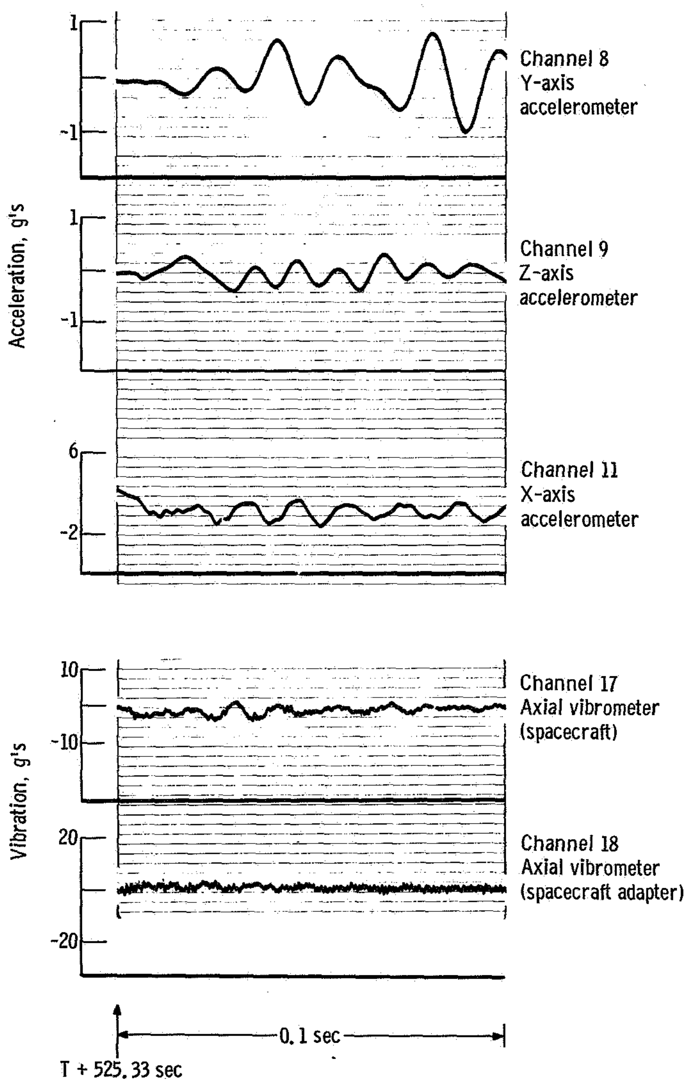


Figure D-10. - Dynamic data near time of Agena engine first cutoff, Mariner Venus 67.

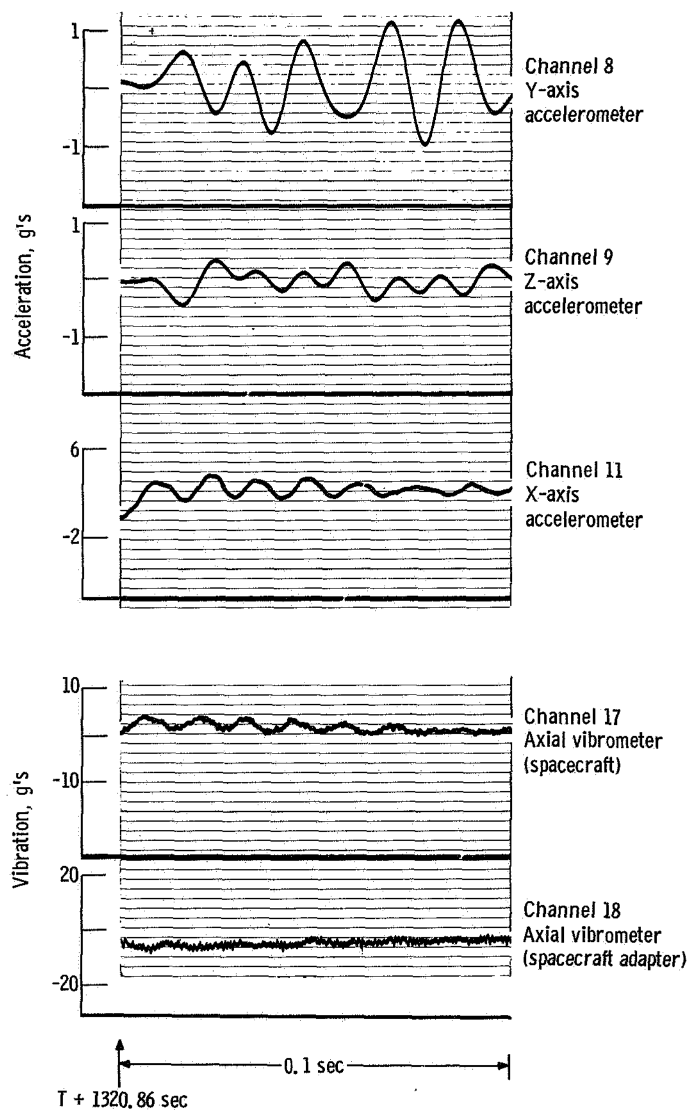


Figure D-11. - Dynamic data near time of Agena engine second ignition, Mariner Venus 67.

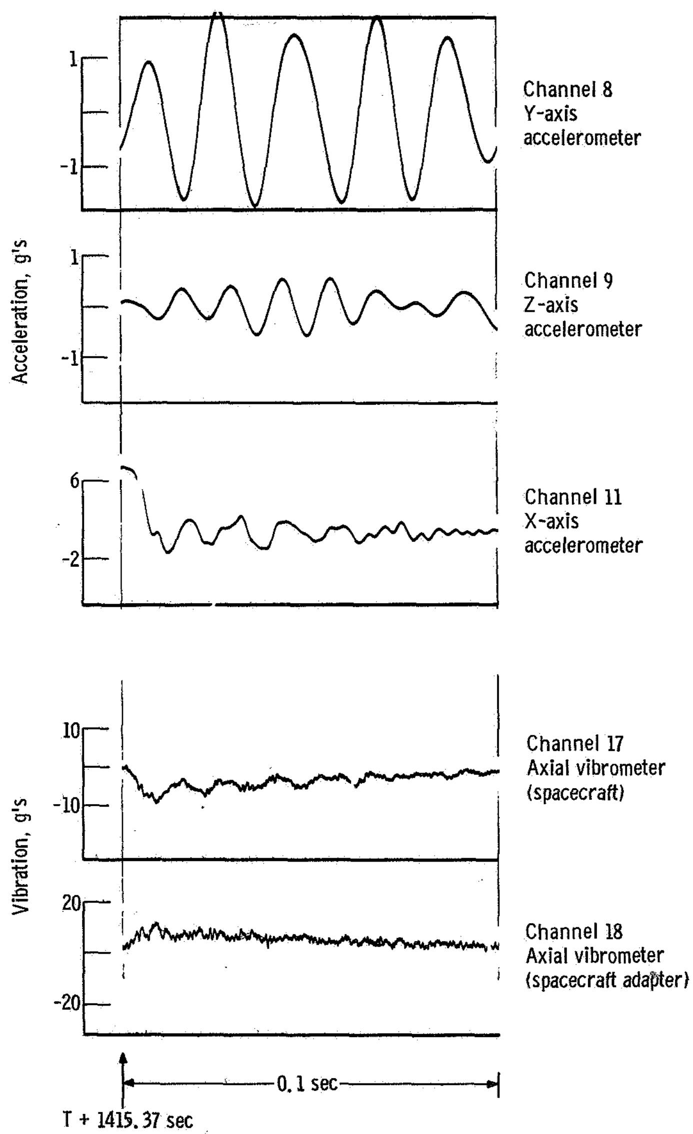


Figure D-12. - Dynamic data near time of Agena engine second cutoff, Mariner Venus 67.

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